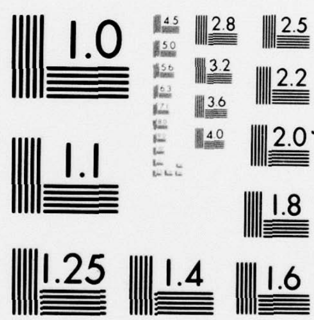
AD-A080 489 BOEING AEROSPACE CO SEATTLE WA BOEING MILITARY AIRPL--ETC F/G 9/2
EASY ACLS DYNAMIC ANALYSIS. VOLUME III. DESCRIPTION OF SIMULATI--ETC(U)
SEP 79 M K WAHI, P R PERKINS, G S DULEBA F33615-77-C-3054

UNCLASSIFIED AFFDL-TR-79-3105-VOL-3 NL

1 OF 4

AD
A080489


1.0
4.5
5.0
5.6
6.3
2.8
3.2
3.6
4.0
2.5
2.2
2.0
1.1
1.8
1.25
1.4
1.6

AFFDL-TR-79-3105
Volume III

LEVEL III

A079803

EASY-ACLS DYNAMIC ANALYSIS
Volume III
Description of Simulations

M. K. WAHI
P. R. PERKINS
J. R. KILNER

BOEING MILITARY AIRPLANE DEVELOPMENT
BOEING AEROSPACE COMPANY
P.O. BOX 3999
SEATTLE, WASHINGTON 98124

SEPTEMER 1979

TECHNICAL REPORT AFFDL-TR-79-3105, Volume III
Final Report for Period April 1977 to June 1979

Approved for public release; distribution unlimited.

AIR FORCE FLIGHT DYNAMICS LABORATORY
AIR FORCE WRIGHT AERONAUTICAL LABORATORIES
AIR FORCE SYSTEMS COMMAND
WRIGHT-PATTERSON AIR FORCE BASE, OHIO 45433

NOTICE

This technical report has been reviewed and is approved for publication.

David L. Fischer

DAVID L. FISCHER, 1LT, USAF
Project Engineer

Howell K. Brewer

HOWELL K. BREWER
Chief, Mechanical Branch
Vehicle Equipment Division

FOR THE COMMANDER

A B Nutt

A. B. NUTT
Director, Vehicle Equipment Division
Air Force Flight Dynamics Laboratories/FEM

AIR FORCE/56780/30 January 1980 — 100

DISCLAIMER NOTICE

THIS DOCUMENT IS BEST QUALITY PRACTICABLE. THE COPY FURNISHED TO DDC CONTAINED A SIGNIFICANT NUMBER OF PAGES WHICH DO NOT REPRODUCE LEGIBLY.

SECURITY CLASSIFICATION OF THIS PAGE (When Data Entered)

19

REPORT DOCUMENTATION PAGE

READ INSTRUCTIONS BEFORE COMPLETING FORM

1. REPORT NUMBER
18 AFFDL-TR-79-3105, Volu[redacted] 3

2. GOVT ACCESSION NO.

3. RECIPIENT'S CATALOG NUMBER

4. TITLE (and Subtitle)
6 EASY ACLS DYNAMIC ANALYSIS. Volume III.
Description of Simulations.

5. TYPE OF REPORT & PERIOD COVERED
FINAL
APRIL 1977 - JUNE 1979

6. PERFORMING ORG. REPORT NUMBER

7. AUTHOR(s)
10 M. K. WAHI, P. R. PERKINS, G. S. Duleba
J. R. KILNER

8. CONTRACT OR GRANT NUMBER(s)
15 F33615-77-C-3054

9. PERFORMING ORGANIZATION NAME AND ADDRESS
BOEING MILITARY AIRPLANE DEVELOPMENT
BOEING AEROSPACE COMPANY
P.O. BOX 3999, SEATTLE, WA 98124

10. PROGRAM ELEMENT, PROJECT, TASK AREA & WORK UNIT NUMBERS
Project 2402, Task 240201,
Work Unit 24020112

11. CONTROLLING OFFICE NAME AND ADDRESS
AIR FORCE FLIGHT DYNAMICS LABORATORY (AFFDL/FEM)
AIR FORCE WRIGHT AERONAUTICAL LABORATORIES
WRIGHT-PATTERSON AFB, OHIO 45433

12. REPORT DATE
11 Sept[redacted]979

13. NUMBER OF PAGES
353
12 354

14. MONITORING AGENCY NAME & ADDRESS (if different from Controlling Office)
16 24Ø2
17 Ø1

15. SECURITY CLASS. (of this report)
UNCLASSIFIED

15a. DECLASSIFICATION/DOWNGRADING SCHEDULE

16. DISTRIBUTION STATEMENT (of this Report)

Approved for public release; distribution unlimited.

17. DISTRIBUTION STATEMENT (of the abstract entered in Block 20, if different from Report)

9 Final rept. 15 Apr 77-1 Jun 79,

18. SUPPLEMENTARY NOTES

19. KEY WORDS (Continue on reverse side if necessary and identify by block number)
EASY
(Air Cushion Landing Systems)
Drop Test
Take off
Free Flight
Landing

20. ABSTRACT (Continue on reverse side if necessary and identify by block number)

This volume contains a detailed description of the development, results, and conclusions of EASY-ACLS dynamic simulations accomplished during this contract. These consist of free flight, drop test, landing approach-touchdown-slideout, and takeoff roll-rotation-climbout of the Jindivik RPV and the XC-8A. Also included as examples are an air bag skid simulation and an arresting gear simulation.

DD FORM 1 JAN 73 1473 EDITION OF 1 NOV 65 IS OBSOLETE

SECURITY CLASSIFICATION OF THIS PAGE (When Data Entered)

410 258

JOB

FOREWORD

This report presents results of work conducted by the Boeing Company, Seattle, Washington, under Air Force Contract F33615-77-C-3054 "Application of the EASY Dynamic Program to the Analysis of Air Cushion Systems on Aircraft" during the period from 15 April 1977 to 1 June 1979. This contract was conducted under the sponsorship of the Air Force Flight Dynamics Laboratory, Wright-Patterson Air Force Base, Ohio with Mr. Peters Skele and Lt. D. L. Fischer as project engineers.

This report is comprised of three volumes.

- Volume I - Component Mathematical Models
- Volume II - Component Computer Programs (Parts I & II)
- Volume III - Description of Simulations

In addition, a User's Manual (Reference 1) has been written to provide a concise reference for day to day usage.

The results presented were developed by the Boeing Aerospace Company. The program managers were A. J. P. Lloyd, H. H. Straub and J. R. Kilner. The principal investigators were M. K. Wahi, G. S. Duleba, J. R. Kilner and P. R. Perkins.

TABLE OF CONTENTS

TABLE OF CONTENTS (CONTINUED)

TABLE OF CONTENTS (CONCLUDED)

LIST OF ILLUSTRATIONS

LIST OF ILLUSTRATIONS (CONTINUED)

LIST OF ILLUSTRATIONS (CONTINUED)

LIST OF ILLUSTRATIONS (CONTINUED)

LIST OF ILLUSTRATIONS (CONTINUED)

LIST OF ILLUSTRATIONS (CONTINUED)

LIST OF ILLUSTRATIONS (CONTINUED)

LIST OF ILLUSTRATIONS (CONTINUED)

LIST OF ILLUSTRATIONS (CONCLUDED)

LIST OF TABLES

SECTION I

INTRODUCTION

The EASY-ACLS Model Generation and Analysis Program is documented in three main volumes. Volume I contains the mathematical models for the standard components of the program library. This includes the derivation of the dynamic equations of motion for aircraft, the derivation of Boeing's inelastic and elastic trunk-cushion landing systems, the implementation of the Foster-Miller inelastic trunk-cushion component, the derivation of an air bag skid system and an arresting gear system, and a discussion of the required inputs and outputs for component models. Volume II contains a detailed description of all the component subroutines, including micro or macro-flowcharts and detailed listings. Volume III, this volume, contains a description of the development, results and conclusions of EASY computer simulations accomplished during this contract. These include landing, takeoff, free flight and drop test simulations. A description of various tasks of the contract follows. A summary document (Reference 1) has also been prepared as a User's Manual.

Task 1 of the contract required the development of aircraft mathematical modeling components consisting of : Equations of Motion, Wind Modeling, Aerodynamic Force and Moment Modeling; and Engine Modeling. Volume I of this report describes the development of these models while Volume II contains the flowcharts and listings.

Task 2 of the contract required modeling the Jindivik RPV in free flight (without controls) and in clean configuration (without trunk) using the components developed in Task 1; and to determine the longitudinal and lateral stability of the vehicle. Volume III, this volume, covers all aspects of Task 2 work. In addition, Section 5 of the User's Manual (Reference 1) also describes this Task 2 simulation as a worked example.

Task 3 of the contract required development of EASY components used in modeling air cushion systems, including air flow system components, fan and ejector components, inelastic and elastic trunk and cushion components, terrain model component, arresting system component, and air bag skid component. Like Task 1, Volumes I and II cover various aspects of work performed under Task 3.

Task 4 of the contract required that the EASY program capabilities be demonstrated by simulating a Jindivik drop test using the models developed in Tasks 1 and 3. It further required that the simulation results be correlated with those of an actual drop test on a recovery trunk (ACRS No. 2). This report, Volume III, covers all aspects of Task 4 simulation/correlation work.

Task 5 of the contract required that the EASY program capabilities be further demonstrated by landing and take off simulations. The approach, touchdown, and slideout of the Jindivik on the recovery trunk along with the roll out, rotation and climbout of the Jindivik on the takeoff trunk using models developed in Tasks 1 and 3. Like Tasks 2 and 4 this report covers all aspects of Task 5 simulations. In addition, simulations to check out the airbag skid component and the arresting system component are also included in this report.

Task 6 of the contract required modeling the XC-8A experimental ACLS aircraft during trim level flight. A "saw tooth" pertubation of the rudder and elevator were made to evaluate the lateral and longitudinal stability of the aircraft. Volume III describes the results of the Task 6 work.

Task 7 of the contract required that the EASY program capabilities be demonstrated by simulating the XC-8A aircraft during takeoff, landing and taxi. Volume III describes the results of the Task 7 work.

Task 8 of the contract required modeling the YC-14 aircraft during an approach configuration. A square impulse deflection of the rudder and elevator were made to evaluate the lateral and longitudinal stability of the aircraft. Volume III describes the results of the Task 8 work.

SECTION II

JINDIVIK FLIGHT SIMULATIONS

2.1 Objectives

The objectives of these Task 2 simulations were:

- o Using components developed under Task 1, simulate Jindivik RPV inflight and out of ground effect
- o Determine vehicle longitudinal and lateral stability using 6 DOF, 4 DOF, 3 DOF, and 2 DOF components
- o Identify computer time savings associated with lower DOF models

2.2 Technical Approach

The simulations described in this section are for a clean configuration vehicle (no trunk). First a 6 DOF rigid body model together with the engine, aerodynamic, and axis transformation modules was simulated. Straight-and-level trim conditions were determined using the optimal controller.

The longitudinal and lateral stabilities were then determined for the free airplane (no controls) by taking the optimal controller out of the loop and inducing perturbations in the pitch and yaw modes. This procedure was repeated with 4 DOF, 3 DOF and 2 DOF rigid body models inorder to determine the computer time savings associated with these lower DOF models.

2.3 6 DOF Model Description and Results

Section 5 of the User's Manual (Reference 1) describes all aspects of this simulation in sufficient details. As a result, only a brief description will be provided here.

The basic modules required to model airplane dynamics are the 4 components VA, OL, DL, DS. VA takes the airplane states and computes aerovariables such as angle-of-attack, sideslip, and dynamic pressure. OL and DL are the longitudinal and lateral-directional aero force and moment computations. DS contains the rigid body dynamics for integrating the aircraft states, and is driven by the aerodynamic forces and moments generated in OL and DL. All other components making up an airplane model are attached to one or more of these components. For example, nonlinear and external forces and torques due to trunk, engine thrust, etc., are inputs to OL or DL. Control systems, on the other hand, require inputs from state variables in DS or aero-variables in VA and determine the control surface settings which are passed to OL or DL. This model structure is demonstrated with a six degree of freedom model of the Jindivik RPV drone.

In addition to the basic airplane modules VA, DL, OL and DS we need to incorporate the component AC where the basic aerodynamic coefficients (XO, ZO, MO, YB, LB, NB) are interpolated from tabular data (one dimensional tables) as functions of angle of attack (α) or of sideslip (β); component ES which is an engine component; and an optimal control module (O.C.) to obtain straight and level trim. Figure 1 shows the model description for the basic Jindivik airplane. The basic airplane and optimal control module schematic diagram generated by the EASY program is shown in Figure 2. Inputs to the optimal controller component include the attitude angles, altitude, true airspeed and some of the velocity states. The desired trim conditions are specified as inputs to this module and error signals are generated which drive the optimal controller outputs. Figure 2 shows the O.C. inputs and the O.C. outputs which regulate engine thrust, as well as the aileron and elevator (Jindivik control surfaces do not include a rudder). The names of output quantities providing connections between various components are given.

The model generation program also specifies a data requirements list. Figure 3 shows this list for the 6 DOF Model.

```
MMAND CARD ---  MODEL DESCRIPTION    SIX DEGREE TEST CASE
MMAND CARD ---  LOCATION=87    VA    INPUTS=DS
MMAND CARD ---  LOCATION=55    AC    INPUTS=VA
MMAND CARD ---  LOCATION=71    OC
MMAND CARD ---  O.C. INPUTS=ALTDS,VT VA,ROLDS,PITDS,YAWDS
MMAND CARD ---  P  DS,Q  DS,R  DS,V  DS,W  DS
MMAND CARD ---  O.C. OUTPUTS=ELEOL,THRES,AILDL
MMAND CARD ---  LOCATION= 1    ES
MMAND CARD ---  LOCATION= 3    OL    INPUTS=VA,ES,AC
MMAND CARD ---  LOCATION=37    DL    INPUTS=OL,VA,AC
MMAND CARD ---  LOCATION=10    DS    INPUTS=OL,DL
MMAND CARD ---  END OF MODEL
MMAND CARD ---  PRINT
```

Figure 1 Jindivik 6DOF Airplane Model

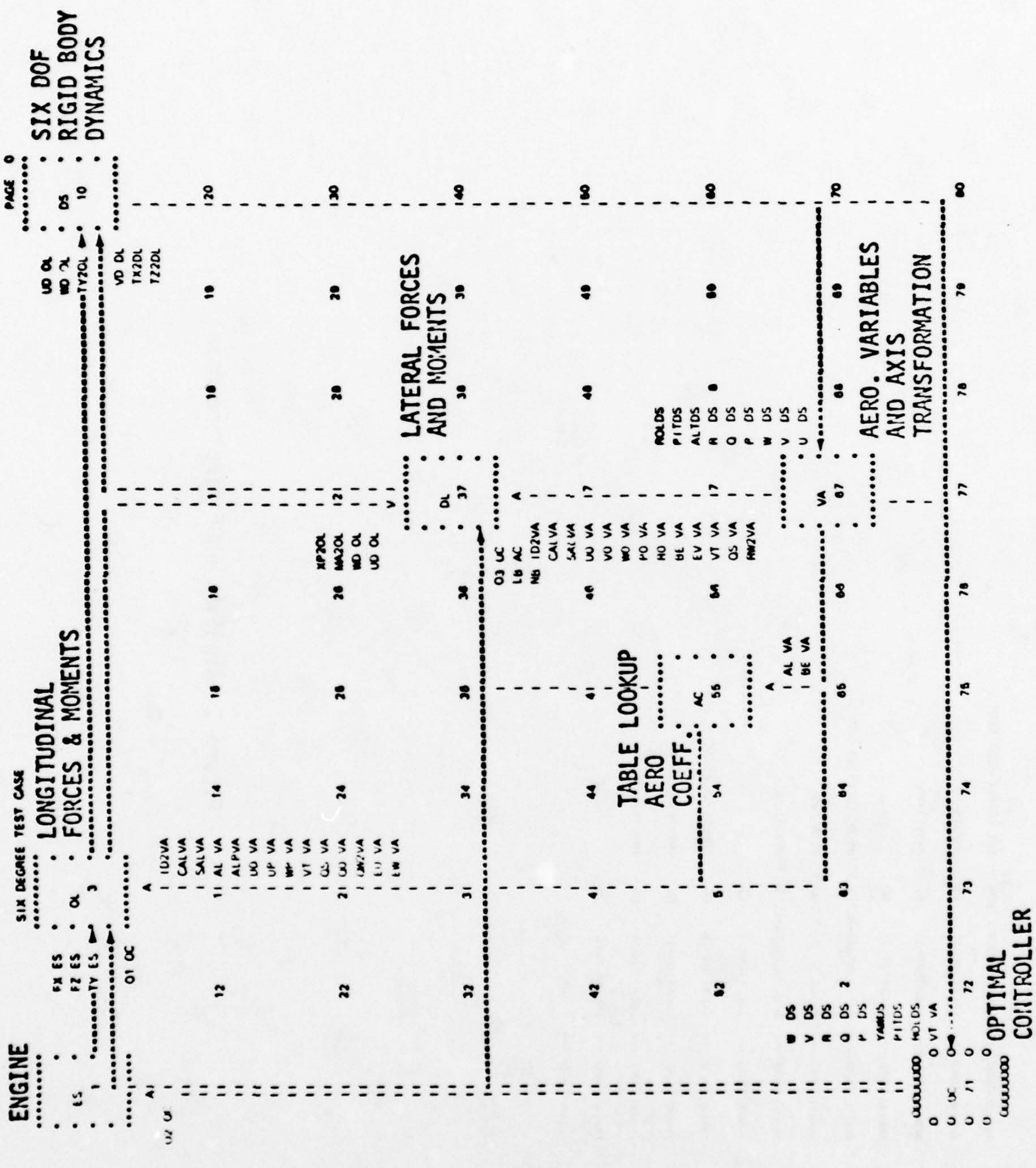


Figure 2 Jindivik Airplane and Trim Components

SIX DEGREE TEST CASE

THIS MODEL CONTAINS 7 COMPONENTS
WITH 12 TABLES 100 PARAMETERS 22 STATES AND 80 VARIABLES

INPUT DATA REQUIREMENTS LIST

TABLES REQUIRED

COMPONENT NAME	TABLE NAME	NO. INDEP. VARIABLES	MAX. DATA ALLOWED
AC	CLTAC	1	30
AC	CDTAC	1	30
AC	CMTAC	1	30
AC	CYTAC	1	30
AC	CLBAC	1	30
AC	CNBAC	1	30
ES	TSRES	2	168
ES	TFNES	2	168
ES	TFPES	2	168
ES	TBTES	1	30
ES	TBPES	1	30
ES	TPOES	1	30

PARAMETERS REQUIRED

COMPONENT NAME	PARAMETER NAME
VA	IDIVA
VA	VS VA
VA	ALSVA
VA	S VA
VA	UW VA
VA	VW VA
VA	WW VA
VA	PW VA
VA	QWIVA
VA	RWIVA
VA	IDGVA
ES	TCOES
ES	AMNES
ES	GAXES
ES	GAZES
ES	XO ES
ES	ZO ES
ES	PAMES
ES	TAMES
ES	PZ ES
ES	IFNES
ES	IBLES
ES	FXIES
OL	XA OL
OL	XU OL
OL	XDEOL
OL	XTROL
OL	XSPOL
OL	XGEOL
OL	KXBOL
OL	ZA OL
OL	ZADOL
OL	ZQ OL
OL	ZU OL
OL	ZDEOL
OL	ZTHOL
OL	ZSPOL
OL	ZGEOL
OL	KZHOL
OL	ZDSOL
OL	MALOL
OL	MADOL
OL	MQ OL
OL	MU OL
OL	MDEOL
OL	MTHOL
OL	MSPOL
OL	MGEOL
OL	KMBOL
OL	MDSOL
OL	MB OL
OL	XGEOL
OL	MATOL
OL	C OL
OL	XPTOL
OL	TSMOL
OL	STAOL
OL	SPOOL
DL	YBDDL
DL	YP DL
DL	YR DL
DL	YDRDL
DL	YDADL
DL	KCYDL
DL	YTHDL
DL	YFSDL
DL	YGEDL
DL	KYBDL
DL	YBRDL
DL	LBDDL
DL	LP DL
DL	LR DL
DL	LDRDL
DL	LDADL
DL	KCLDL
DL	LTHDL
DL	LFSDL
DL	LGEDL
DL	KLBDL
DL	LBRDL
DL	NBDDL
DL	NP DL
DL	NR DL
DL	NDRDL
DL	NDADL
DL	NTHDL
DL	NFSDL
DL	NGEDL
DL	KNBDL
DL	NBRDL
DL	RUDDL
DL	FSPDL
DL	FYTDL
DL	TXTDL
DL	TZTDL
DL	B DL
DS	IXXDS
DS	IYYDS
DS	IZZDS
DS	IXZDS

STATES
(INITIAL CONDITIONS AND ERROR CONTROLS REQUIRED)

COMPONENT NAME	STATE NAME
OC	X1 OC
OC	X2 OC
OC	X3 OC
OC	X4 OC
OC	X5 OC
OC	X6 OC
OC	X7 OC
OC	X8 OC
OC	X9 OC
OC	X10OC
OC	X11OC
ES	TH ES
DS	U DS
DS	V DS
DS	W DS
DS	P DS
DS	Q DS
DS	R DS
DS	ROLDS
DS	PITDS
DS	YAWDS
DS	ALTDS

Figure 3 Model Input Data Requirements List

The analysis data used to demonstrate this model is shown in Figure 4. Assumptions made for the analysis are as follows:

Aircraft weight = 2700 lbs.

Moments of Inertia

I_{xx} = 1190 slug ft^2

I_{yy} = 1811 slug ft^2

I_{zz} = 2840 slug ft^2

I_{xz} = -200 slug ft^2

Engine moment arms

x axis arm = 10.9 ft

z axis arm = .375 ft

Flight Altitude = 30,000 ft

Aircraft speed = 430 knots

Following the command PARAMETER VALUES are the input data values for the aircraft components DL, VA, OL, ES, and DS. The aero coefficient tables are then input, and the non-zero INITIAL CONDITIONS for states complete the model definition. See Sections 4.1, 4.2 and 4.3 of Users Manual for a description of the input format corresponding to these commands. The aerodynamic tabular data was obtained from Reference 2 for 4^{o} flaps while the supplementary data for stability derivatives was taken from Reference 3.

The commands beginning with O.C. DATA and ending with STEADY STATE, design the optimal controller, do a linear analysis and find the trim conditions (see Section 5.2 of the Users Manual for an explanation of O.C. design and trim calculations). Figure 5 shows the printout of O.C. DESIGN-LINEAR ANALYSIS results. Figure 6 shows the printout of the STEADY STATE analysis for this case. The significant non-zero trim conditions are boxed. The results give a steady state trim which approximates the desired flight conditions. If a more accurate solution is desired we can redesign the optimal controller at the new operating point and repeat the trim procedure, or put a heavier weight on the conditions requiring greater accuracy and rerun the analysis.

```
PARAMETER VALUES
YP DL=-.0215,LP DL=-.370,NP DL=.00167,LR DL=.4498 }
NR DL=-.1545,LDADL=-.14184,NDADL=.0152            }---- LATERAL DERIVATIVES
IDGVA=6,ISWOL=3,IBLES=1,IFNES=1
ID1VA=3,VS VA=726,ALSVA=0,S  VA=76
TCOES=1,GAXES=1,GAZES=0,XO ES=-10.9,ZO ES=-.375,FX1ES=0.---- ENGINE DATA
IXXDS=1190,IYYDS=1811,IZZDS=2840,IXZDS=-200
C  OL=4,B  DL=19,MA1OL=83.92,XP1OL=C
MDEOL=.642,MQ OL=-.246,ZDEOL=.2594              LONGITUDINAL DERIVATIVES
TABLE,CLTAC,7 ---------------------------------------
-15,-10,-5,0,5,10,15
-1.27,-.758,-.243,.272,.787,1.3,1.82
TABLE,CDTAC,12
-1.0,-.5,-.1,-.05,0,.05,.1,.15,.2,.5,1.0,1.5
.0775,.0565,.0397,.0382,.0373,.0368,.0368,.0368,.0375,.0488,.0675,.0863
TABLE,CMTAC,10
-1.0,-.5,-.1,0,.1,.2,.3,.5,1.0,1.5
.134,.094,.0615,.0535,.0465,.04,.034,.022,-.008,-.038      AERO DYNAMIC
TABLE.CYTAC,7                                             COEFFICIENTS
-90,-75,-60,0,60,75,90                                    AS TABULAR
-.75,-.75,-.75,-.75,-.75,-.75,-.75                        INPUTS
TABLE,CNBAC,3
-20,0,20
.125,.08,.035
TABLE,CLBAC,3
-20,0,20
.188,-.072,-.332 ------------------------------------
INITIAL CONDITIONS
U  DS=726,ALTDS=30000,PITDS=1,TH ES=377
O.C. DATA ----------------------------
YOP=30000,726,0,1,0,0,0,0,0,0
UOP=0,377,0
Q=1,100,100,1,100,1,1,1,1,1
RU=.01,.01,.01                          O.C. DESIGN
PRINT CONTROL=3                         AND TRIM
DESIGN O.C.
LINEAR ANALYSIS
STEADY STATE ------------------------
XIC-X
PRINTER PLOTS,PLOT ON
PLOT ID=     M.K.WAHI      M/S  47-03
PLOT ALL TABLES
DISPLAY1
AL VA,VS,WW VA
PITDS,VS,WW VA
FX2OL,VS,WW VA
FZ2OL,VS,WW VA
TY2OL,VS,WW VA
TITLE      STEADY SCAN,LONGITUDINAL MOTION
SS PARAMETER=WW VA,SS START=40,SS STOP=0
SS POINTS=5,STEADY STATE
SS PARAMETER=NONE
DISPLAY1
AL VA,VS,TIME
W  DS,VS,TIME
U  DS,VS,TIME
Q  DS,VS,TIME
PITDS,VS,TIME
DISPLAY2
ALTDS,VS,TIME
```

Figure 4 Analysis Program-Input Data and Commands

```
TINC=.1,TMAX=12,PRATE=1C
NO STATES
INT CCNTROL=W  DS=1,U  DS=1,Q  DS=1,PITDS=1,ALTDS=1
TITLE   FREE LONGITUDINAL MODEL,NONDIM. DATA
LINEAR ANALYSIS
INITIAL CONDITION=W  DS=60
SIMULATE
DISPLAY1
BE VA,VS,VW VA
TX2CL,VS,VW VA
TZ2CL,VS,VW VA
FY2CL,VS,VW VA
TITLE     STEADY SCAN, LATERAL MOTION
INITIAL CONDITION=W  DS=-16.43
SS PARAMETER=VW VA,SS START=40,SS STOP=0
SS POINTS=5,STEADY STATE
SS PARAMETER=NONE
DISPLAY1
BE VA,VS,TIME
R  CS,VS,TIME
P  DS,VS,TIME
ROLCS,VS,TIME
ALTCS,VS,TIME
TINC=.1,TMAX=12,PRATE=10
NO STATES
INT CONTROL=V  DS=1,P  DS=1,ROLDS=1,R  DS=1
TITLE     FREE LATERAL MODEL (STABILITY AXIS DATA)
LINEAR ANALYSIS
INITIAL CONDITICN=V  DS=16.3,W  DS=-16.43
SIMULATE
DISPLAY2
AL VA,VS,TIME
W  CS,VS,TIME
U  CS,VS,TIME
Q  CS,VS,TIME
PITCS,VS,TIME
NO STATES
INT CCNTROL=U  DS=1,V  DS=1,W  DS=1,P  DS=1
Q  DS=1,R  DS=1,PITDS=1,ROLDS=1,ALTDS=1
TITLE     FREE LONGI-LATERAL MODEL(STABILITY AXIS DATA)
LINEAR ANALYSIS
INITIAL CONDITIONS
V  CS=16.3,W  DS=60
SIMULATE
TITLE      LONGITUDINAL TORQUE TRANSFER FUNCTION
IF INPUT=TY20L,TF OUTPUT=PITDS,TRANSFER FUNCTION
NICHOLS,TRANSFER FUNCTION
NYQUIST,TRANSFER FUNCTION
```

Figure 4 Analysis Program-Input Data and Commands (Concluded)

			22 EIGENVALUES		
	REAL		IMAGINARY	NATURAL FREQ.	DAMPING RATIO
1	-.441029E-01		0.	.441029E-01	1.00000
2	-.331057	+-	2.89225	2.91113	.113721
3	-.761057	+-	.614752	.978329	.777915
4	-1.00020		0.	1.00020	1.00000
5	-1.21515	+-	.772786	1.44006	.843816
6	-6.66369		0.	6.66369	1.00000
7	-10.0724		0.	10.0724	1.00000
8	-10.1134		0.	10.1134	1.00000
9	-10.1247		0.	10.1247	1.00000
10	-10.2259		0.	10.2259	1.00000
11	-12.7877		0.	12.7877	1.00000
12	-13.4285		0.	13.4285	1.00000
13	-13.6496		0.	13.6498	1.00000
14	-48.9593		0.	48.9593	1.00000
15	134.423		0.	134.423	1.00000
16	136.891		0.	136.891	1.00000
17	-145.555		0.	145.566	1.00000
18	-252.338		0.	252.338	1.00000
19	-413.783		0.	413.783	1.00000

All negative real eigenvalues indicate a stable system.

.333000 CPU SECONDS WERE REQUIRED FOR THE PREVIOUS ANALYSIS

COMMAND CARD ----- STEADY STATE ← COMMAND FOR NEXT ANALYSIS (TRIM)

Figure 5 Linear Analysis Output of O.C. Design

/*/*/*/ STEADY STATE ANALYSIS /*/*/*/

A MAXIMUM OF 30 ITERATIONS CAN BE USED

TIME = 0.

STATES

1 X1 OC	= 206.20	2 X2 OC	= -4.5936	3 X3 OC	= -.48667E-04	4 X4 OC	= -18.742	5 X5 OC	= .17174E-04
6 X6 OC	= -5.6826	7 X7 OC	= -.40810E-04	8 X8 OC	= .43895E-06	9 X9 OC	= -2.8132	10 X10OC	= .16035E-03
11 X11OC	= 28.497	12 $\boxed{\text{TH ES = 583.48}}$		13 $\boxed{\text{U DS = 720.41}}$		14 V DS	= .70572E-18	15 $\boxed{\text{W DS = -18.427}}$	
16 P DS	= .70941E-21	17 Q DS	= .45343E-18	18 R DS	= .93841E-23	19 ROLDS	= .29995E-18	20 $\boxed{\text{PITDS = -1.3082}}$	
21 YAWDS	= .16562E-03	22 $\boxed{\text{ALTDS = 30027.}}$							

RATES

1 R1	= .19546E-06	2 R2	= .16621E-04	3 R3	= .22010E-07	4 R4	= -.13290E-08	5 R5	= -.19271E-08
6 R6	= .17314E-06	7 R7	= -.10029E-09	8 R8	= .64051E-11	9 R9	= -.83375E-08	10 R10	= -.94961E-11
11 R11	= -.69907E-07	12 R12	= .36780E-11	13 R13	= -.71070E-06	14 R14	= .14661E-19	15 R15	= .32087E-06
16 R16	= .56159E-14	17 R17	= .95257E-06	18 R18	= -.64231E-15	19 R19	= .70920E-21	20 R20	= .45343E-18
21 R21	= .93865E-23	22 R22	= -.34129E-05						

AILERON DEFL.

VARIABLES

1 UO VA	= 720.41	2 VO VA	= .70572E-18	3 WO VA	= -18.427	4 PO VA	= .70941E-21	5 QO VA	= .45343E-18
6 RO VA	= .93841E-23	7 ID2VA	= 3.0000	8 QW2VA	= 0.	9 HW2VA	= 0.	10 CALVA	= 1.0000
11 [illegible]ALVA	= 0.	12 $\boxed{\text{AL VA = -1.3082}}$		13 ALPVA	= -1.3082	14 VT VA	= 720.69	15 BE VA	= .66113E-18
16 W[illegible] VA	= 18.427	17 UP VA	= -.77055E-02	18 EQ VA	= .73346	19 EV VA	= .16807E-18	20 EW VA	= 37.166
21 [illegible]VA	= .37375	22 QC VA	= 262.50	23 QS VA	= 17528.	24 MACVA	= .72437	25 ZO AC	= -.13746
26 [illegible] AC	= -.36800E-01	27 MD AC	= .44065E-01	28 YB AC	= -.75000	29 LB AC	= -.65019E-01	30 NB AC	= .82939E-01
31 $\boxed{\text{[illegible] OC = -2.6541}}$		32 $\boxed{\text{D2 OC = 583.48}}$		33 $\boxed{\text{D3 OC = -.13878E-15}}$		34 FX ES	= 583.48	35 FZ ES	= 0.
36 TV ES	= -218.61	37 FSPES	= -1	38 FSTES	= -1	39 PPUES	= -1	40 TPUES	= -1
41 W2 ES	= -1	42 T2 ES	= -1	43 FX2OL	= -61.552	44 FZ2OL	= -2699.3	45 TY2OL	= .30109E-03
46 UD OL	= -.71070E-05	47 WD OL	= .32087E-06	48 MA2OL	= 83.920	49 XP2OL	= 0.	50 FY2DL	= -.12875E-18
51 VD DL	= .14651E-19	52 TX2DL	= .11440E-12	53 TZ2DL	= -.12234E-13	54 XD DS	= 720.69	55 YD DS	= .20830E-02
56 PD DS	= .56159E-14	57 QD DS	= .95257E-05	58 RD DS	= -.64231E-16	59 HODDS	= .70920E-21	60 PIDDS	= .45343E-18

ELEVATOR DEFLECTION

ENGINE THRUST

PARAMETERS

1 ID1VA	= 3.0000	2 VS VA	= 728.00	3 ALSVA	= 0.	4 S VA	= 78.000	5 UW VA	= 0.
6 VW VA	= 0.	7 WW VA	= 0.	8 PW VA	= 0.	9 QW1VA	= 0.	10 RW1VA	= 0.
11 I[illegible]VA	= 6.0000	12 TCOES	= 1.0000	13 AMNES	= 0.	14 GAXES	= 1.0000	15 GAZES	= 0.
16 [illegible] ES	= -10.900	17 ZO ES	= -.37500	18 PAMES	= 14.700	19 TAMES	= 469.00	20 P2 ES	= .99999
21 [illegible]NES	= 1.0000	22 IBLES	= 1.0000	23 FATES	= 0.	24 XA OL	= 0.	25 XU OL	= 0.
26 [illegible]OL	= 0.	27 XTHOL	= 0.	28 XSPOL	= 0.	29 XGEOL	= 0.	30 KXBOL	= 1.0000
31 ZA OL	= 0.	32 ZADOL	= 0.	33 ZQ OL	= 0.	34 ZU OL	= 0.	35 ZDEOL	= .25940
36 ZTHOL	= 0.	37 ZSPOL	= 0.	38 ZGEOL	= 0.	39 KZBOL	= 1.0000	40 ZDSOL	= 0.
41 MA[illegible]OL	= 0.	42 MADOL	= 0.	43 MQ OL	= -.24600	44 MU OL	= 0.	45 MDEOL	= .84200
46 MTHOL	= 0.	47 MSPOL	= 0.	48 MGEOL	= 0.	49 KMBOL	= 1.0000	50 MDSOL	= 0.
51 M[illegible] OL	= 0.	52 KGEOL	= 0.	53 MATOL	= 83.920	54 C OL	= 4.0000	55 XP1OL	= 0.
56 [illegible]OL	= 3.0000	57 STAOL	= 0.	58 SPDOL	= 0.	59 YBDDL	= 0.	60 YP DL	= -.21800E-01
61 [illegible] DL	= 0.	62 YRHDL	= 0.	63 YDADL	= 0.	64 KCYDL	= 1.0000	65 YTHDL	= 0.
66 [illegible]DL	= 0.	67 YGEDL	= 1.0000	68 KYBDL	= 1.0000	69 YBRDL	= 0.	70 LBDDL	= 0.
71 LP DL	= -.37000	72 LR DL	= .44980	73 LDRDL	= 0.	74 LDADL	= -.14184	75 KCLDL	= 1.0000
76 LTHDL	= 0.	77 LFSDL	= 0.	78 LGEDL	= 1.0000	79 KLBDL	= 1.0000	80 LBRDL	= 0.
81 NBDDL	= 0.	82 NP DL	= .16700E-02	83 NR DL	= -.15450	84 NDRDL	= 0.	85 NDADL	= .15200E-01
86 NTHDL	= 0.	87 NFSDL	= 0.	88 NGEDL	= 1.0000	89 KNBDL	= 1.0000	90 NBRDL	= 0.
91 RUDDL	= 0.	92 FSPDL	= 0.	93 FY1DL	= 0.	94 TX1DL	= 0.	95 TZ1DL	= 0.
96 B DL	= 19.000	97 IXXDS	= 1190.0	98 IYYDS	= 1811.0	99 IZZDS	= 2840.0	100 IXZDS	= -200.00

SYSTEM EIGENVALUES AT THIS OPERATING POINT

22 EIGENVALUES

Figure 6 Trim Condition Determined by the O.C.

Note that the state derivatives (RATES) are all driven to very small values in Figure 6. After the steady state analysis, the XIC-X command causes the trim values of the states to replace the original initial conditions.

The next command PRINTER PLOTS causes the on line printer to be turned on while the PLOT ON part of the command requests the offline printer e.g. Calcomp or SC4020 to be turned on. The PLOT TABLES command causes an off line plotting of the named tables. The commands beginning with DISPLAY1 and ending with SS PARAMETER=NONE causes a STEADY SCAN analysis for a down wind velocity component of 0 to 40 ft/sec. The off line plotted results of the STEADY SCAN analysis are shown in Figure 7.

All states except the longitudinal states are then frozen in order to simulate the free longitudinal aircraft, see Figure 8. This command also removes the optimal controller from the model by freezing its states. The optimal controller states will remain frozen since trim has been determined. The commands starting with DISPLAY1 and ending with ALTDS, VS, TIME specify the time histories to be plotted. The command LINEAR ANALYSIS gives the pole locations for the free airplane, and the commands TINC = .1 plus the command SIMULATE produce the time history simulations. (Note that the perturbation is specified by setting the inital condition W DS = 60).

The results of the LINEAR ANALYSIS are shown in Figure 9. The time history plots of alpha, w, u, q and pitch for this case are shown in Figure 10. The short period transient dies out after about 4 seconds, while u and pitch oscillate slowly at the phugoid mode. Figure 11 shows the corresponding printer plot output for AL VA (angle of attack).

Another STEADY SCAN analysis is requested for a side wind velocity parameter (VW VA). This is followed by a LINEAR ANALYSIS and a SIMULATION of the free lateral airplane where all states except the lateral states are frozen out (for commands see Figure 4). The results of the LINEAR ANALYSIS are shown in Figure 12 while the time history plots of beta, r, p, altitude and roll are shown in Figure 13. Note that roll angle settles to a 1.30 degrees offset due to the spiral divergence mode.

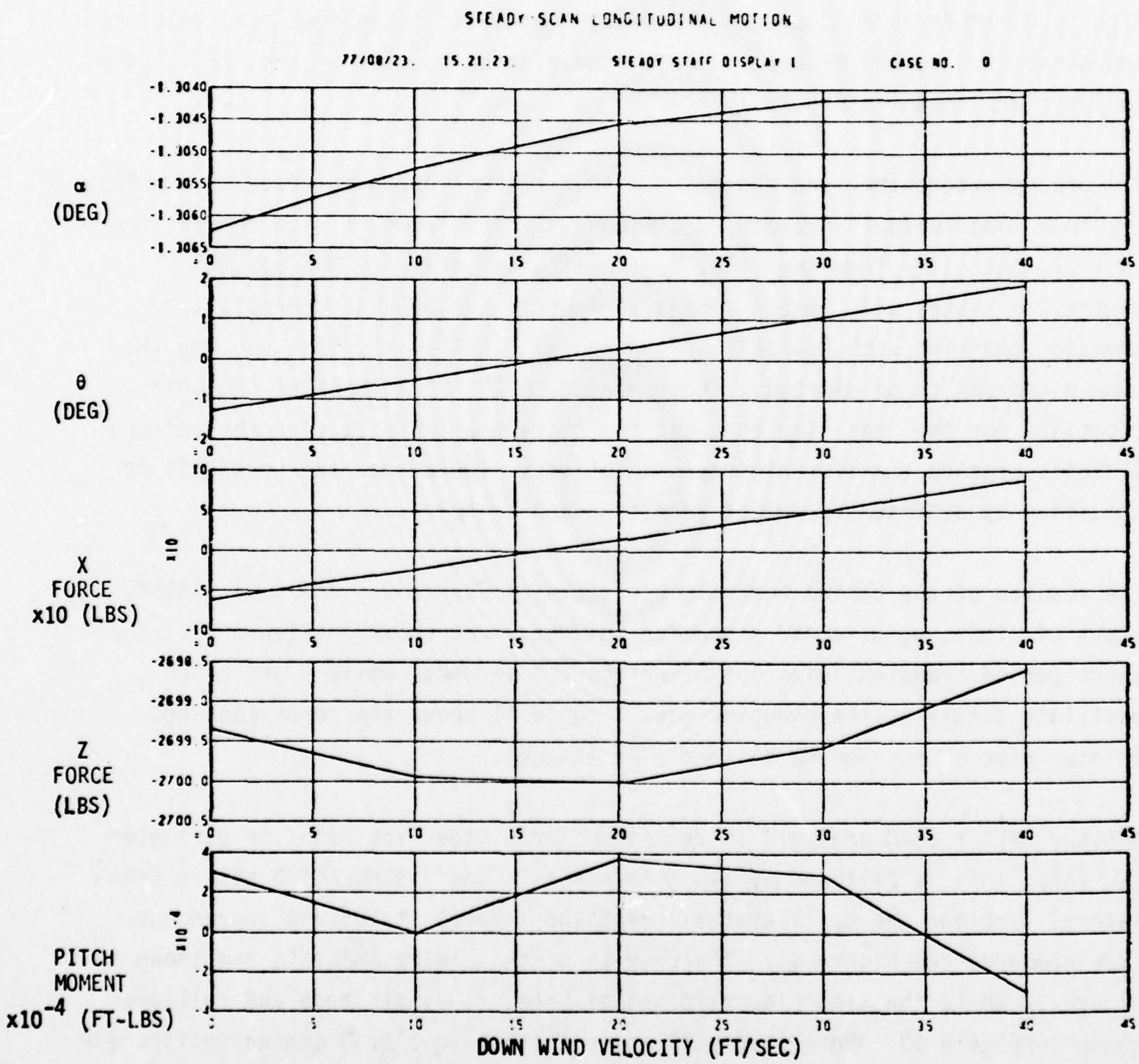


Figure 7 Offline Plots of Down-Wind Steady Scan

16 XO ES	= -10.900	17 ZO ES	= -.37600	18 PAMES	= 14.700	19 TAMES	= 480.00	20 P2 ES	= .[illegible]
21 IENF	= 1.0000	22 IBLES	= 1.0000	23 FX1ES	= 0.	24 XA OL	= 0.	25 XU OL	= 0.
26 XDEL	= 0.	27 XTROL	= 0.	28 XSPOL	= 0.	29 XGEOL	= 0.	30 KXBOL	= 1.0000
31 ZA OL	= 0.	32 ZADOL	= 0.	33 ZQ OL	= 0.	34 ZU OL	= 0.	35 ZDEOL	= .25940
36 ZTHOL	= 0.	37 ZSPOL	= 0.	38 ZGEOL	= 0.	39 KZBOL	= 1.0000	40 ZDSOL	= 0.
41 MALOL	= 0.	42 MADOL	= 0.	43 MQ OL	= -.24800	44 MU OL	= 0.	45 MDEOL	= .64200
46 MTHOL	= 0.	47 MSPOL	= 0.	48 MGEOL	= 0.	49 KMBOL	= 1.0000	50 MDSOL	= 0.
51 M[illegible] OL	= 0.	52 KGEOL	= 0.	53 MATOL	= 83.920	54 C OL	= 4.0000	55 XP1OL	= 0.
56 [illegible]	= 3.0000	57 STAOL	= 0.	58 SPOOL	= 0.	59 YBDOL	= 0.	60 YP DL	= -.21500E-01
61 [illegible]H DL	= 0.	62 YDRDL	= 0.	63 YDADL	= 0.	64 KCYDL	= 1.0000	65 YTRDL	= 0.
66 Y[illegible] DL	= 0.	67 YGEDL	= 1.0000	68 KYBDL	= 1.0000	69 YBHDL	= 0.	70 LBDDL	= 0.
71 LP DL	= -.37000	72 LR DL	= .44980	73 LDRDL	= 0.	74 LDADL	= -.14184	75 KCLDL	= 1.0000
76 LTHDL	= 0.	77 LFSDL	= 0.	78 LGEDL	= 1.0000	79 KLBDL	= 1.0000	80 LBHDL	= 0.
81 N[illegible]DL	= 0.	82 NP DL	= .16700E-02	83 NR DL	= -.15450	84 NDRDL	= 0.	85 NDADL	= .15200E-01
86 NTHDL	= 0.	87 NFSDL	= 0.	88 NGEDL	= 1.0000	89 KNBDL	= 1.0000	90 NBHDL	= 0.
91 [illegible]DL	= 0.	92 FSPDL	= 0.	93 FY1DL	= 0.	94 TX1DL	= 0.	95 TZ1DL	= 0.
96 B DL	= 19.000	97 IXXDS	= 1180.0	98 IYYDS	= 1811.0	99 IZZDS	= 2840.0	100 IXZDS	= -200.00

END OF STEADY SCAN ANALYSIS

.833000 CPU SECONDS WERE REQUIRED FOR THE PREVIOUS ANALYSIS

COMMAND CARD ----- SS PARAMETER=NONE ←——TURNING OFF STEADY SCAN

COMMAND CARD ----- DISPLAY1

COMMAND CARD ----- AL VA,VS,TIME

COMMAND CARD ----- W DS,VS,TIME

COMMAND CARD ----- U DS,VS,TIME

COMMAND CARD ----- Q DS,VS,TIME

COMMAND CARD ----- PITDS,VS,TIME

COMMAND CARD ----- DISPLAY2

COMMAND CARD ----- ALTDS,VS,TIME

} REPLACING STEADY SCAN PARAMETERS WITH VARIABLES FOR TIME HISTORY SIMULATION

COMMAND CARD ----- TINC=.1,TMAX=12,PRATE=10 ←—— REQUESTING PRINTOUT EVERY (.1*10=1.) SECOND

COMMAND CARD ----- NO STATES ←—— Freezing all states; Taking O.C. out

COMMAND CARD ----- INT CONTROL-W DS=1,U DS=1,Q DS=1,PITDS=1,ALTDS=1 ←—— Unfreezing Longitudinal states

COMMAND CARD ----- TITLE FREE LONGITUDINAL MODEL,NONDIM. DATA

COMMAND CARD ----- LINEAR ANALYSIS ←—— Next Analysis

Figure 8 Playback of Input Commends for next Analysis

/*/*/*/ LINEAR ANALYSIS /*/*/*/

STATE NAME	OPERATING POINT	PERTURBATION SIZE	INTEGRATOR CONTROL
1 X1 OC	206.20	.100	0
2 X2 OC	-4.5936	.100	0
3 X3 OC	-.46557E-04	.100	0
4 X4 OC	-16.742	.100	0
5 X5 OC	.17174E-04	.100	0
6 X6 OC	-5.6826	.100	0
7 X7 OC	-.40810E-04	.100	0
8 X8 OC	.43895E-05	.100	0
9 X9 OC	-2.5132	.100	0
10 X10OC	.16035E-03	.100	0
11 X11OC	26.497	.100	0
12 TH ES	583.48	.100	0
13 U DS	720.41	.100	1
14 V DS	.70572E-18	.100	0
15 W DS	-16.427	.100	1
16 P DS	.70941E-21	.100	0
17 Q DS	.45343E-18	.100	1
18 R DS	.93841E-23	.100	0
19 ROLDS	.29995E-18	.100	0
20 PITDS	-1.3062	.100	1
21 YAWDS	.16562E-03	.100	0
22 ALTDS	30027.	.100	1

RATES AT OPERATING POINT

1 X1 OC	= .19546E-08	2 X2 OC	= .18621E-04	3 X3 OC	= .22010E-07	4 X4 OC	= -.13290E-08	5 X5 OC	= -.10271E-
6 X6 OC	= .17334E-08	7 X7 OC	= -.10029E-09	8 X8 OC	= .64051E-11	9 X9 OC	= -.83375E-08	10 X10OC	= -.949812-
11 X11OC	= -.69907E-07	12 TH ES	= .36380E-11	13 U DS	= -.71070E-08	14 V DS	= .14661E-19	15 W DS	= .32087E-
16 P DS	= .56159E-14	17 Q DS	= .95257E-05	18 R DS	= -.64231E-15	19 ROLDS	= .70920E-21	20 PITDS	= .46343E-
21 YAWDS	= .93865E-23	22 ALTDS	= -.34129E-06						

STABILITY MATRIX

	U DS	W DS	Q DS	PITDS	ALTDS
U DS	-.2133E-01	.4648E-03	.2867	-.5614	.2718E-03
W DS	-.1283	-1.708	12.57	.1231E-01	.1138E-02
Q DS	-.7712E-02	-1.181	-.2843E-01	0.	-.2445E-03
PITDS	0.	0.	1.000	0.	0.
ALTDS	-.2280E-01	-.9887	0.	12.68	0.

5 EIGENVALUES

	REAL	IMAGINARY	NATURAL FREQ.	DAMPING RATIO
1	.319681E-07	0.	.319681E-07	-1.00000
2	.928587E-02	+- .814770E-01	.820044E-01	.113236
3	-.868640	+- 3.76040	3.86942	.225070

Phugoid; t = 77 sec

SHORT PERIOD; t=1.63 sec

$$t = \frac{2\pi}{\text{NATURAL FREQ.}}$$

.490000E-01 CPU SECONDS WERE REQUIRED FOR THE PREVIOUS ANALYSIS

Figure 9 Jindivik Free Longitudinal Linear Analysis

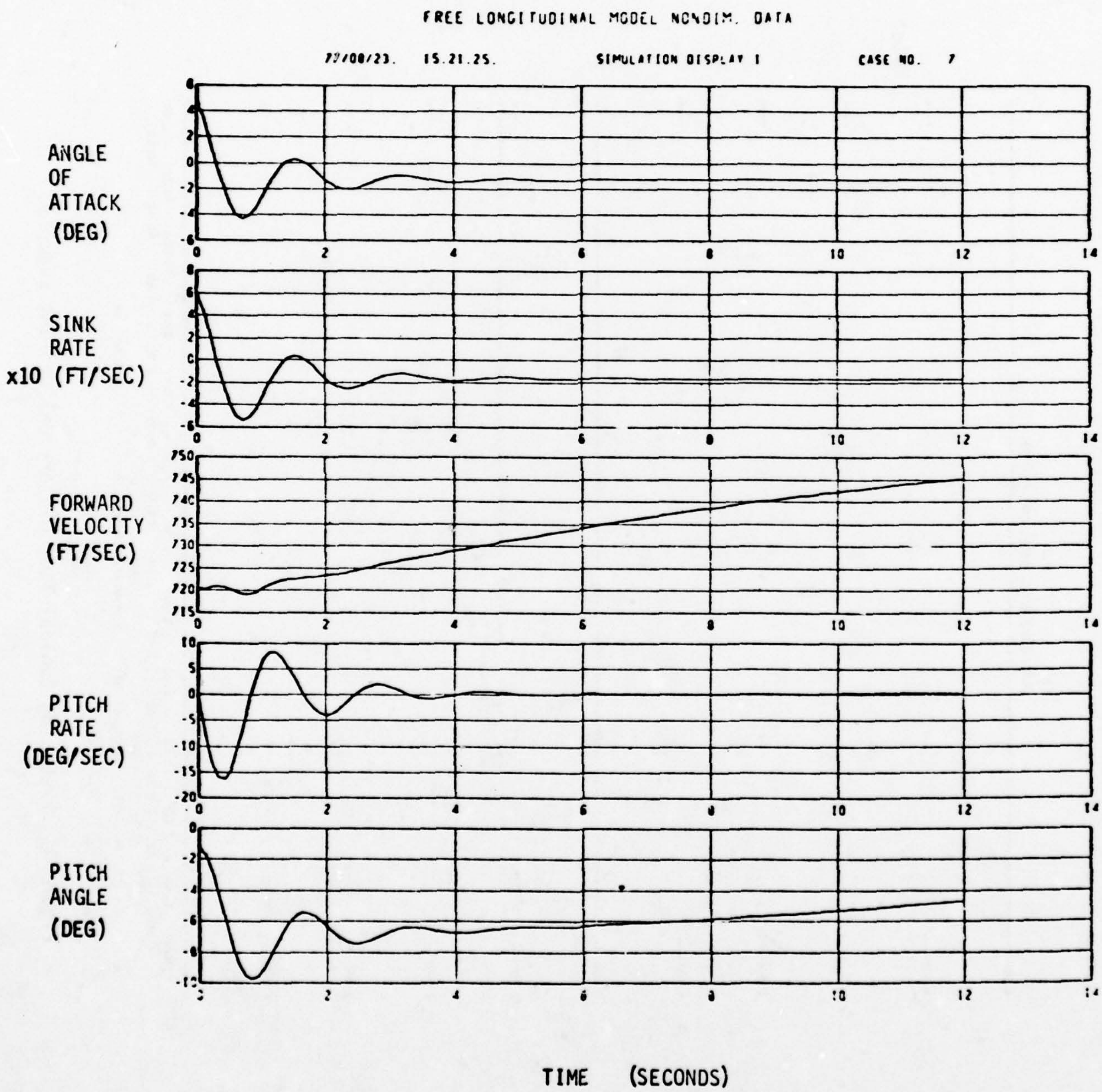
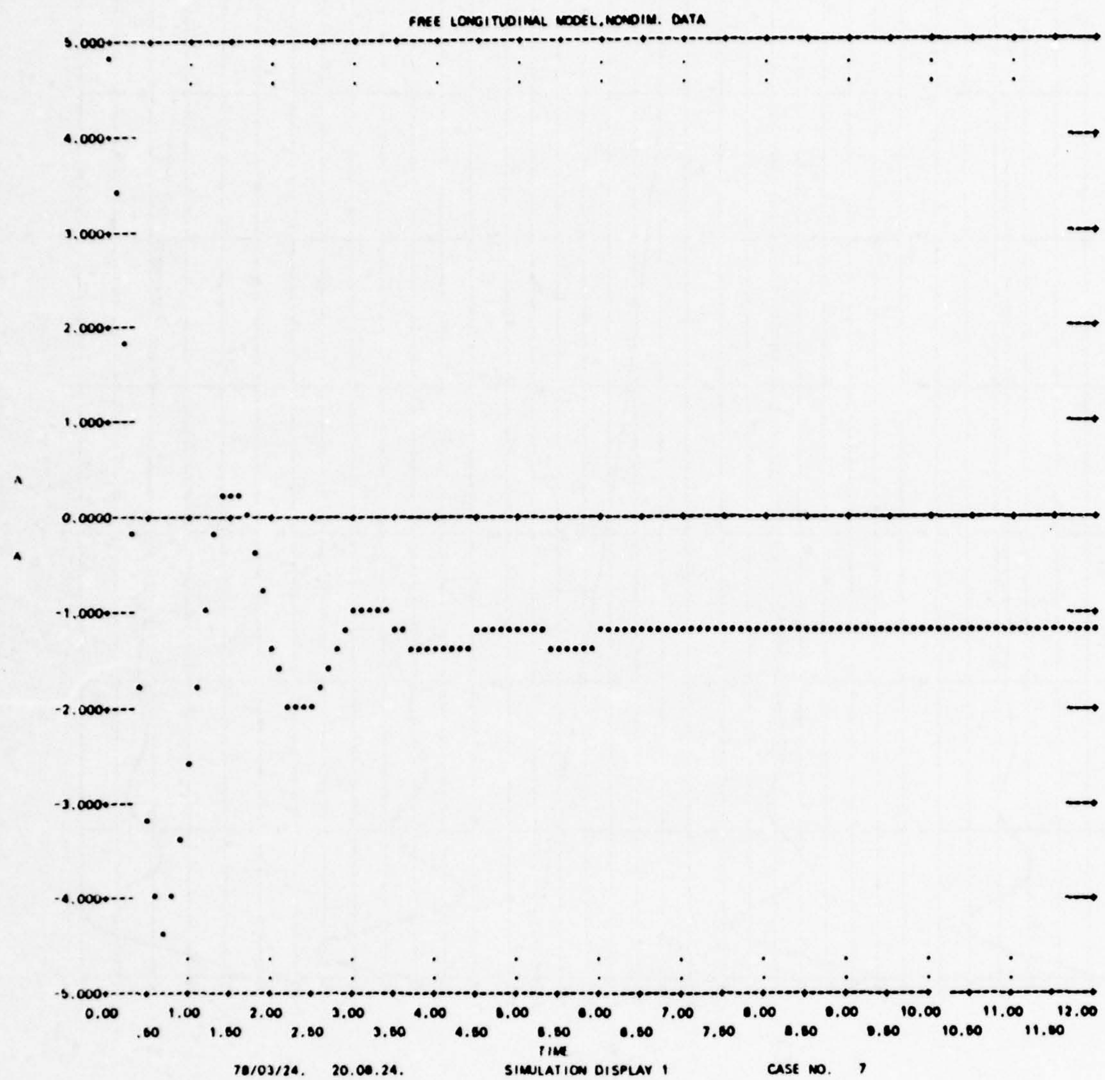


Figure 10 Jindivik Free Lingitudinal Time History Plots



Note: In actual listing five more plots will follow; see commands on Figure 4

Figure 11 Free Lingitudinal Time History Online Plot Example

/*/*/*/ LINEAR ANALYSIS /*/*/*/

STATE NAME	OPERATING POINT	PERTURBATION SIZE	INTEGRATOR CONTROL
1 X1 OC	206.20	.100	0
2 X2 OC	-4.5936	.100	0
3 X3 OC	-.48557E-04	.100	0
4 X4 OC	-16.742	.100	0
5 X5 OC	.17174E-04	.100	0
6 X6 OC	-5.6826	.100	0
7 X7 OC	.40810E-04	.100	0
8 X8 OC	.43855E-05	.100	0
9 X9 OC	-2.5132	.100	0
10 X10OC	.16035E-03	.100	0
11 X11OC	26.497	.100	0
12 TH ES	583.48	.100	0
13 U DS	720.41	.100	0
14 V DS	.70572E-18	.100	1
15 W DS	-16.430	.100	0
16 P DS	.70341E-21	.100	1
17 Q DS	.45143E-18	.100	0
18 R DS	.93841E-23	.100	1
19 HOLDS	.29995E-18	.100	1
20 PITDS	-1.3062	.100	0
21 YAWDS	.16562E-03	.100	0
22 ALTDS	30027.	.100	0

RATES AT OPERATING POINT

1 X1 OC	= .16543E-04	2 X2 OC	= .88602E-03	3 X3 OC	= .88718E-08	4 X4 OC	= -.44870	5 X5 OC	= -.20690E-
6 X6 OC	= -.26528E-03	7 X7 OC	= .19615E-07	8 X8 OC	= .98204E-07	9 X9 OC	= -.17730E-03	10 X10OC	= .13417E-
11 X11OC	= .15730E-02	12 TH ES	= .36380E-11	13 U DS	= -.86281E-06	14 V DS	= .14661E-10	15 W DS	= .83386E-
16 P DS	= .56159E-14	17 Q DS	= .36982E-02	18 R DS	= -.64231E-15	19 HOLDS	= .70920E-21	20 PITDS	= .45343E-
21 YAWDS	= .93865E-23	22 ALTDS	= .31193E-02						

STABILITY MATRIX

	V DS	P DS	R DS	ROLDS
V DS	-.2174	-.2878	-12.57	.5814
P DS	-1.370	-1.382	1.720	0.
R DS	.8698	.9990E-01	-.3600	0.
ROLDS	0.	1.000	-.2280E-01	-.1805E-21

4 EIGENVALUES

	REAL	IMAGINARY	NATURAL FREQ.	DAMPING RATIO	
1	.387503E-01	0.	.387503E-01	-1.00000	SPIRAL DIVERGENCE
2	-.286514	+- 3.22841	3.24110	.884003E-01	DUTCH ROLL
3	-1.42303	0.	1.42303	1.00000	ROLL SUBSIDENCE

.41000E-01 CPU SECONDS WERE REQUIRED FOR THE PREVIOUS ANALYSIS

Figure 12 Jindivik Free Lateral Linear Analysis

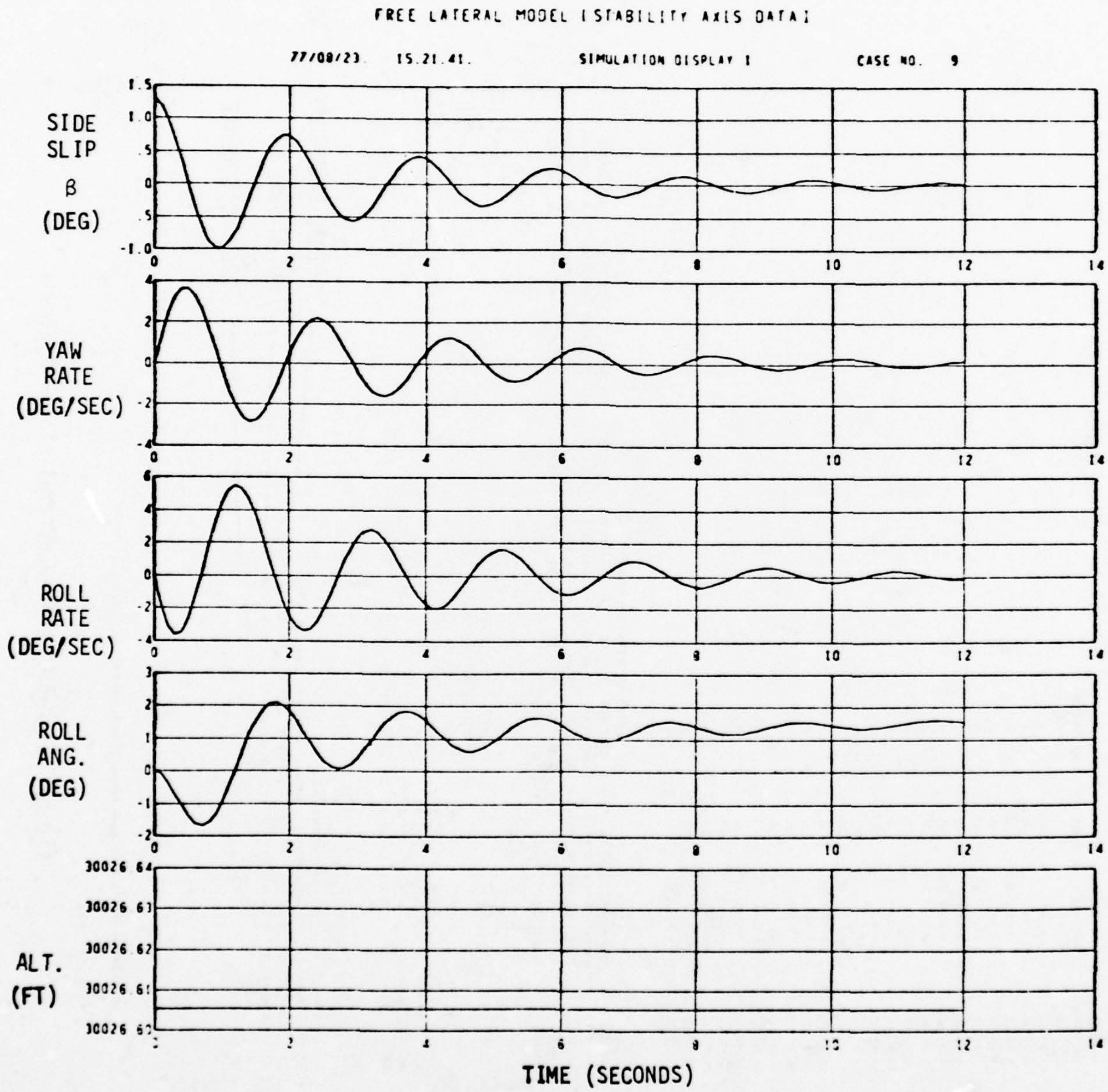


Figure 13 Jindivik Free Lateral Time History Plots

Referring back to Figure 4, we now ask for a LINEAR ANALYSIS and a SIMULATION of the free airplane with all longitudinal and lateral states (except yaw) active. The results of the LINEAR ANALYSIS are shown in Figure 14. Note that the 9 eigenvalues match the 5 eigenvalues of the Figure 9 plus the 4 eigenvalues of Figure 12 as expected. Similarly the time history plots of the SIMULATION for this case, match those of Figures 10 and 12. The time history plots and the associated linear analysis of Figures 10 and 13 indicate that the vehicle (Jindivik) is basically stable in both longitudinal and lateral modes.

The next command was for a TRANSFER FUNCTION analysis of the pitch moment vs pitch angle and Figure 15 shows the off line BODE plots of analysis results. Note that the phugoid and the short period modes of oscillations can be detected in Figure 15 (also see Figure 9).

2.4 Lower DOF Models

The EASY program has the capability of reducing the order (degrees of freedom) of a system by "freezing" state variables. This is a very useful technique, especially for troubleshooting an unstable system. However, if repeated analyses with reduced degrees of freedom are anticipated, components specifically tailored to the required degrees of freedom should be used. This results in computational efficiency and clarity of output. It was with this intent that the lower order models of 4 DOF, 3 DOF-lateral, 3 DOF-longitudinal and 2 DOF-longitudinal were designed under Task 1 and tested under Task 2.

The trim and longitudinal stability analyses of 6 DOF (Section 2.3) were repeated with a 3 DOF longitudinal component. Figures 16 through 20 show the inputs and outputs for the 3 DOF model corresponding respectively to Figures 1, 2, 3, 6 and 10 for the 6 DOF model. Identical trim conditions and time-history plots have been obtained without any loss in accuracy and an associated computer time savings of 20% as shown in Table 1. Since an airplane trim is determined in the longitudinal mode only (i.e. no control surface deflections for lateral controls) the use of 3 DOF model to determine trim is valid and recommended.

/*/*/*/ LINEAR ANALYSIS /*/*/*/

STATE NAME	OPERATING POINT	PERTURBATION SIZE	INTEGRATOR CONTROL
1 X1 OC	206.20	.100	0
2 X2 OC	-4.5936	.100	0
3 X3 OC	-.48557E-04	.100	0
4 X4 OC	-16.742	.100	0
5 X5 OC	.17174E-04	.100	0
6 X6 OC	-5.6626	.100	0
7 X7 OC	-.40810E-04	.100	0
8 X8 OC	.43695E-05	.100	0
9 X9 OC	-2.5132	.100	0
10 X10OC	.16035E-03	.100	0
11 X11OC	26.497	.100	0
12 TH ES	583.48	.100	0
13 U DS	720.41	.100	1
14 V DS	16.300	.100	1
15 W DS	-16.430	.100	1
16 P DS	.70941E-21	.100	1
17 Q DS	.45143E-18	.100	1
18 R DS	.93841E-23	.100	1
19 ROLDS	.29995E-18	.100	1
20 PITDS	-1.3062	.100	1
21 YAWDS	.16562E-03	.100	0
22 ALTDS	30027.	.100	1

6 EIGENVALUES

	REAL	IMAGINARY	NATURAL FREQ.	DAMPING RATIO
1	.387764E-01	0.	.387764E-01	-1.00000
2	.748928E-07	0.	.748928E-07	-1.00000
3	-.929652E-02	+- .814874E-01	.820159E-01	.113350
4	-.287163	+- 3.22776	3.24051	.886167E-01
5	-.863294	+- 3.74911	3.84722	.224394
6	-1.43398	0.	1.43398	1.00000

RATES AT OPERATING POINT

1 X1 OC	= .14365E-01	2 X2 OC	= 1.2134	3 X3 OC	= 2180.8	4 X4 OC	= -.46988	5 X5 OC	= -21.813
6 X6 OC	= .12804E-01	7 X7 OC	= -6.1341	8 X8 OC	= .59078	9 X9 OC	= -.81400E-01	10 X10OC	= -.80258
11 X11OC	= -.38071E-02	12 TH ES	= .38380E-11	13 U DS	= -.39416E-02	14 V DS	= -3.6447	15 W DS	= -.11117E-
16 P DS	= -22.345	17 Q DS	= .72422E-02	18 R DS	= 14.183	19 ROLDS	= .70920E-21	20 PITDS	= .48343E-
21 YAWDS	= .93865E-23	22 ALTDS	= .31193E-02						

4 ELEMENTS OF /RATIO/ DIFFER FROM 1 BY 10/. THESE ELEMENTS ARE PRECEDED BY AN * IN THE STABILITY MATRIX

RATIO(3, 7) = .500000
RATIO(5, 4) = .500000
RATIO(5, 6) = .500000
RATIO(8, 7) = .611583

STABILITY MATRIX

	U DS	V DS	W DS	P DS	Q DS	R DS	ROLDS	PITDS	ALTDS
U DS	-.2133E-01	-.4840E-03	.4849E-03	0.	.2888	.2848	0.	-.5614	.2718E-03
V DS	-.4814E-02	-.2176	.1117E-03	-.2878	0.	-12.57	.5614	0.	.1252E-( )
W DS	-.1283	-.2025E-02	-1.709	-.2846	12.57	0.	*-.4899E-03	.1231E-01	.1136E-02
P DS	-.3553E-01	-1.372	-.3741	-1.382	0.	1.721	0.	0.	.7892E-03
Q DS	-.7721E-02	.4362E-03	-1.182	* .1927E-03	-.2844E-01	*-.1927E-03	0.	0.	-.2447E-03
R DS	.1965E-01	.8707	-.1244E-02	.9993E-01	0.	-.3601	0.	0.	-.5009E-03
ROLDS	0.	0.	0.	1.000	-.1194E-23	-.2280E-01	-.1808E-21	.1839E-24	0.
PITDS	0.	0.	0.	0.	1.000	-.5235E-20	*-.7070E-23	0.	0.
ALTDS	-.2280E-01	0.	-.9997	0.	0.	0.	-.2847	12.58	0.

Figure 14 Jindivik Free Longi-Lat. Linear Analysis

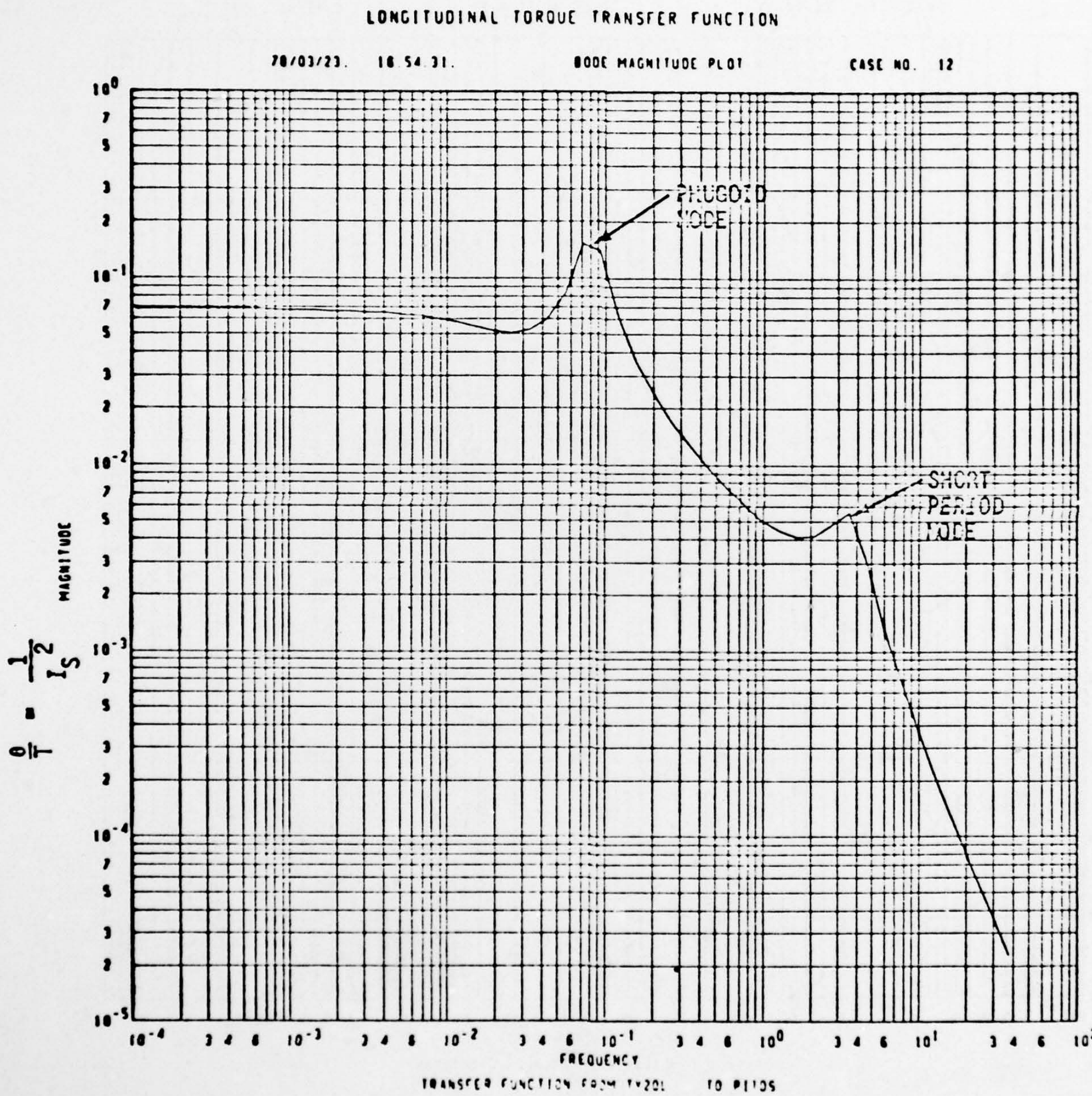


Figure 15 Transfer Function Output "BODE" Plot

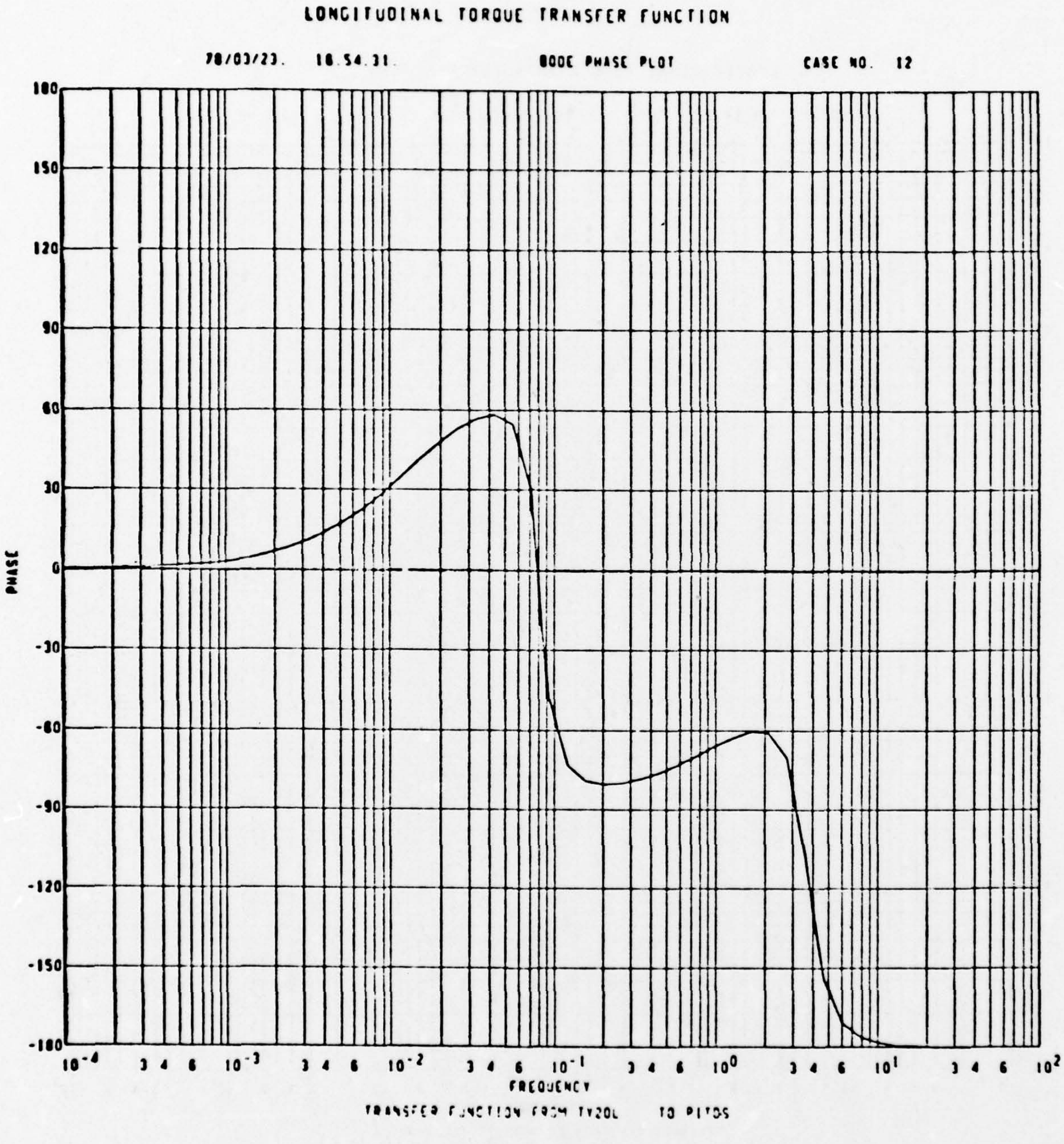


Figure 15 Transfer Function Output "Bode" Plot (Concluded)

```
MODEL DESCRIPTION     THREE DOF LONGITUDINAL TEST CASE
LOCATION=67     VA    INPUTS=TL
LOCATION=55     AC    INPUTS=VA
LOCATION=71     OC
O.C. INPUTS=ALTTL,VT VA,PITTL
Q  TL,W  TL
O.C. OUTPUTS=ELEOL,THRES
LOCATION=1      ES
LOCATION=3      OL   INPUTS=VA,ES,AC
LOCATION=10     TL    INPUTS=OL
ENC CF MODEL
PRINT
```

Figure 16 Jindivik 3DOF (Longitudinal) Airplane Model

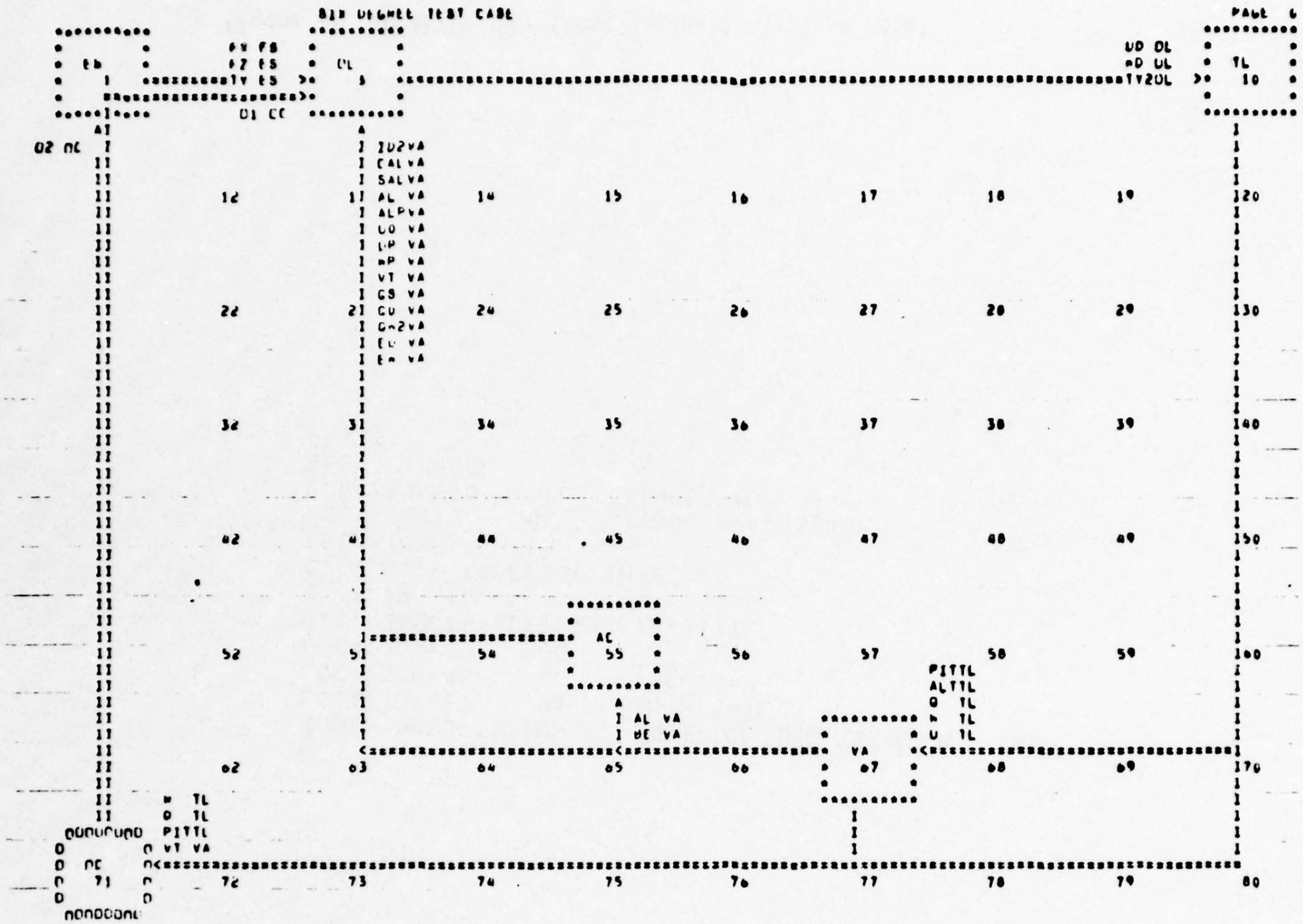


Figure 17 Jindivik 3DOF Airplane and Trim Components

```
PARAMETER VALUES
IDGVA=3,IBLES=1,IFNES=1,ISWOL=3
IDIVA=3,VS VA=726,ALSVA=0,S  VA=76
TCOES=1,GAXES=1,GAZES=0,XO ES=-10.9,ZO ES=-.375,FX1ES=0
IYYTL=1811,ROLTL=0,YAWTL=0
C  OL=4,MA1OL=83.92,XP1OL=0
MDEOL=.642,MQ OL=-.246,ZDEOL=.2594
TABLE,CLTAC,7
-15,-10,-5,0,5,10,15
-1.27,-.758,-.243,.272,.787,1.3,1.82
TABLE,CDTAC,12
-1.0,-.5,-.1,-.05,0,.05,.1,.15,.2,.5,1.0,1.5
.0775,.0565,.0397,.0382,.0373,.0368,.0368,.0368,.0375,.0488,.0675,.0863
TABLE,CMTAC,10
-1.0,-.5,-.1,0,.1,.2,.3,.5,1.0,1.5
.134,.094,.0615,.0535,.0465,.04,.034,.022,-.008,-.038
INITIAL CONDITIONS
U  TL=726,ALTTL=30000,PITTL=1,TH ES=377
O.C. DATA
YOP=30000,726,1,0,0
UOP=0,377
Q=1,100,1,1,1
RU=.01,.01
PRINT CONTROL=3
DESIGN O.C.
LINEAR ANALYSIS
STEADY STATE
XIC-X
PRINTER PLOTS,PLOT ON
PLOT ID=     M.K.WAHI      M/S 47-03
PLOT ALL TABLES
DISPLAY1
AL VA,VS,TIME
W  TL,VS,TIME
U  TL,VS,TIME
Q  TL,VS,TIME
PITTL,VS,TIME
DISPLAY2
ALTTL,VS,TIME
TINC=.1,TMAX=12,PRATE=10
NO STATES
INT CONTROL=W  TL=1,U  TL=1,Q  TL=1,PITTL=1,ALTTL=1
TITLE      FREE LONGITUDINAL MODEL,NONDIM. DATA
LINEAR ANALYSIS
INITIAL CONDITIONS=W  TL=60
SIMULATE
```

Figure 18 3DOF Analysis Program Data and Commands

/*/*/*/ STEADY STATE ANALYSIS /*/*/*/

A MAXIMUM OF 30 ITERATIONS CAN BE USED

TIME = 0.

STATES

1 X1 DC	206.18	2 X2 DC	-4.5936	3 X3 DC	-16.745	4 X4 DC	-5.6843	5 X5 DC	-2.5136
6 X6 DC	26.504	7 TH ES	583.47	8 U TL	720.41	9 W TL	-16.430	10 Q TL	.10173E-17
11 PITTL	-1.3065	12 ALTTL	30027.						

RATES

1 R1	.19571E-06	2 R2	.16541E-04	3 R3	-.13309E-06	4 R4	.17363E-06	5 R5	-.83482E-06
6 R6	-.69544E-07	7 R7	0.	8 R8	-.71143E-05	9 R9	.32051E-05	10 R10	.95724E-05
11 R11	.10173E-17	12 R12	-.34062E-05						

VARIABLES

1 UO VA	720.41	2 VO VA	0.	3 WO VA	-16.430	4 PO VA	0.	5 QO VA	.10173E-17
6 RO VA	0.	7 ID2VA	3.0000	8 JW2VA	0.	9 RW2VA	0.	10 CALVA	1.0000
11 SALVA	0.	12 AL VA	-1.3065	13 ALPVA	-1.3065	14 VT VA	720.59	15 BE VA	0.
16 WP VA	-16.430	17 UP VA	-.77056E-02	18 EU VA	.73358	19 EV VA	.56130	20 EW VA	32.161
21 SIGVA	.37375	22 QC VA	262.50	23 QS VA	17528.	24 MACVA	.72437	25 ZO AC	-.13743
26 XO AC	-.35800E-01	27 MU AC	.44067E-01	28 YB AC	2.0000	29 LB AC	2.0000	30 NB AC	2.0000
31 D1 DC	-3.6543	32 D2 DC	583.47	33 FX ES	583.47	34 FZ ES	0.	35 TY ES	-216.60
36 FSPES	0.	37 FSTES	0.	38 PPUES	0.	39 TPUES	0.	40 PPDES	0.
41 TPDES	0.	42 FX2OL	-61.563	43 FZ2OL	-2698.9	44 TY2OL	.30256E-03	45 UD DL	-.71143E-05
46 WD DL	.32051E-05	47 MA2OL	83.920	48 XP2OL	0.	49 XD TL	720.59	50 QD TL	.95724E-05

PARAMETERS

1 V VA	0.	2 P VA	0.	3 R VA	0.	4 ROLVA	.99999	5 ID1VA	3.0000
6 VS VA	726.00	7 ALSVA	0.	8 S VA	76.000	9 JW VA	0.	10 VW VA	0.
11 WW VA	0.	12 PW VA	0.	13 QW1VA	0.	14 RW1VA	0.	15 IDGVA	3.0000
16 YCOES	1.0000	17 AMNES	0.	18 GAXES	1.0000	19 GAZES	0.	20 XO ES	-10.900
21 ZO ES	-.37500	22 PAMES	14.700	23 TAMES	459.00	24 W ES	0.	25 IFNES	1.0000
26 IBLES	1.0000	27 XA DL	0.	28 XU DL	0.	29 XDEDL	0.	30 XTRDL	0.
31 XSPDL	0.	32 XGEDL	0.	33 KXBDL	1.0000	34 ZA DL	0.	35 ZADDL	0.
36 ZQ DL	0.	37 ZU DL	0.	38 ZDEDL	.25940	39 ZTRDL	0.	40 ZSPDL	0.
41 ZGEDL	0.	42 KZBDL	1.0000	43 ZDSDL	0.	44 MALDL	0.	45 MADDL	0.
46 MQ DL	-.24600	47 MU DL	0.	48 MDEDL	.64200	49 MTRDL	0.	50 MSPDL	0.
51 MGEDL	0.	52 KMBDL	1.0000	53 MDSDL	0.	54 MB DL	0.	55 KGEDL	0.
56 MA1DL	83.920	57 C DL	4.0000	58 XP1DL	0.	59 ISWDL	3.0000	60 STADL	0.
61 SPODL	0.	62 IYYTL	1611.0	63 ROLTL	0.	64 YAWTL	0.		

SYSTEM EIGENVALUES AT THIS OPERATING POINT

12 EIGENVALUES

	REAL	IMAGINARY	NATURAL FREQ.	DAMPING RATIO
1	-.670761	+- 1.15301	1.33393	.502847
2	-.978646	+- .530649E-01	.980084	.998533
3	-1.01117	0.	1.01117	1.00000
4	-7.25838	0.	7.25838	1.00000
5	-10.8838	+- 2.54628	11.1777	.973708
6	-14.5020	0.	14.5020	1.00000
7	-136.854	0.	136.854	1.00000
8	-145.551	0.	145.551	1.00000
9	-252.691	0.	252.691	1.00000

Figure 19 Trim Conditions Determined by the O.C. (3DOF)

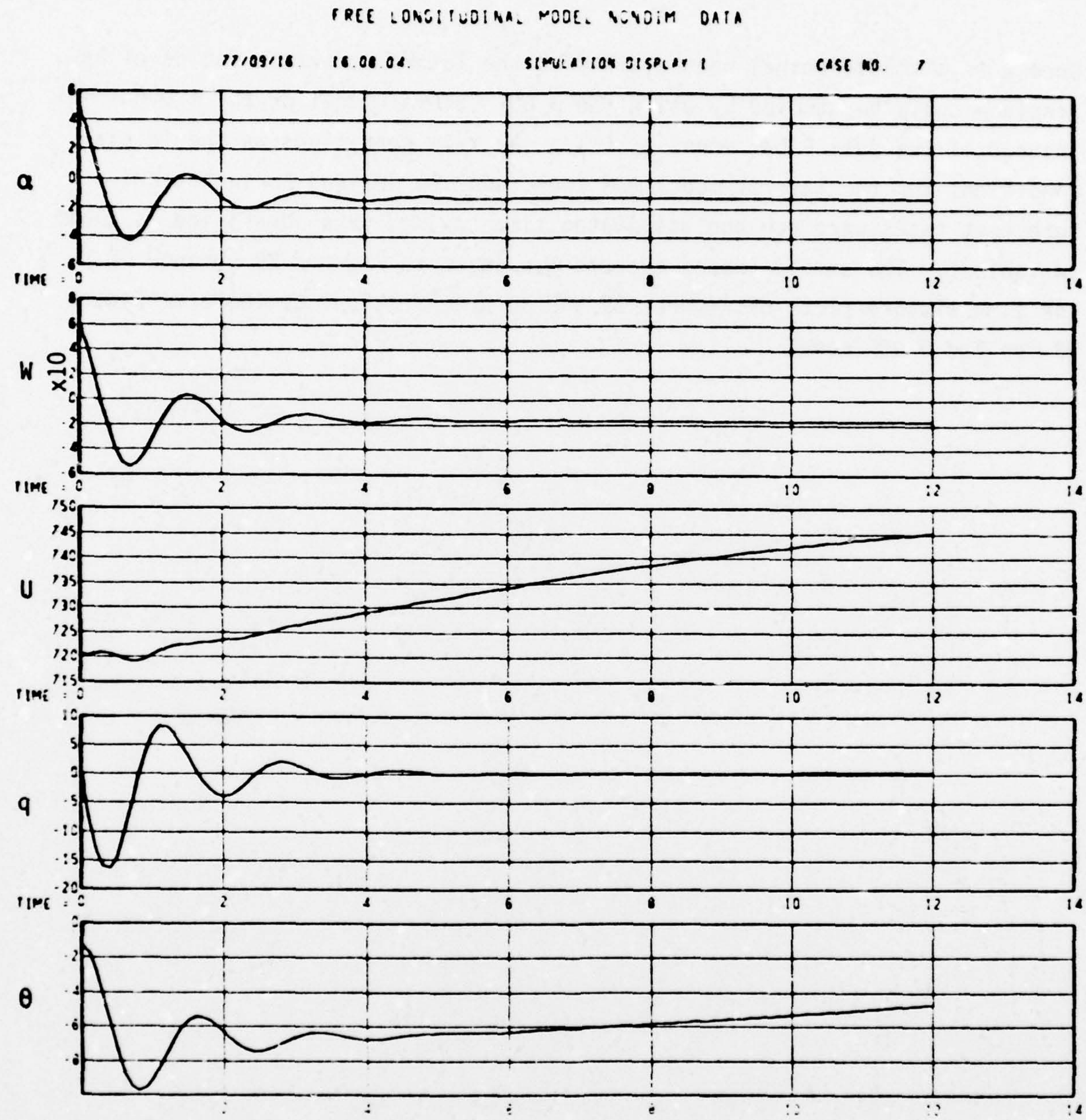


Figure 20 Jindivik Free Longitudinal Time History Plots (3DOF)

These same analysis were repeated using the 2 DOF model and the corresponding inputs-outputs are shown in Figures 21 through 25. The associated time savings are shown in Table 1.

Once a trim condition has been determined the lateral characteristics of an airplane could be studied by using the 3 DOF lateral model or the 4 DOF model instead of the full 6 DOF model by using the trim conditions as the initial conditions for the lateral model and then inducing desired perturbations. Such test cases were run and associated time savings were identified as shown in Table 1. The sample inputs-outputs are shown in Figures 26 through 33. The time history plots of Figures 29 and 33 are identical to those of Figure 13 for the 6 DOF model.

TABLE 1 TASK 2 COMPUTER USUAGE FOR VARIOUS DOF MODELS

TYPE OF RUN	COMPUTER RESOURCES UNITS (CRU'S)			PERCENT DIFFERENCE
	6DOF	3DOF	DIFFERENCE	
TRIM CALCULATIONS	4.057	3.425	0.632	16
	3.724	2.947	0.727	20
TRIM + SIMULATION (10 SEC)	5.781	4.227	1.554	27
SIMULATION + TF (12 SEC)	9.522	6.644	2.878	30
	9.315	6.129	3.186	34
SIMULATION (F106) (LATERAL)	6.045 (3D FROZEN)	5.108	0.937	16
SIMULATION (LATERAL)	(4DOF) 3.40 (1D FROZEN)	(3DOF) 3.40	0	0
(LONGITUDINAL) TRIM + SIMULATION	(3DOF) 5.346	(2DOF) 4.721	0.625	12

```
MODEL DESCRIPTION     TWO DEGREE TEST CASE
LOCATION=67      VA    INPUTS=TT
LOCATION=55      AC    INPUTS=VA
LOCATION=71      OC
O.C. INPUTS=ALTTT,VT VA,PITTT
Q  TT,W  TT
O.C. OUTPUTS=ELEOL,THRES
LOCATION=1      ES
LOCATION=3      OL    INPUTS=VA,ES,AC
LOCATION=10      TT    INPUTS=OL
END OF MODEL
PRINT
```

Figure 21 Jindivik 2DOF Longitudinal Model

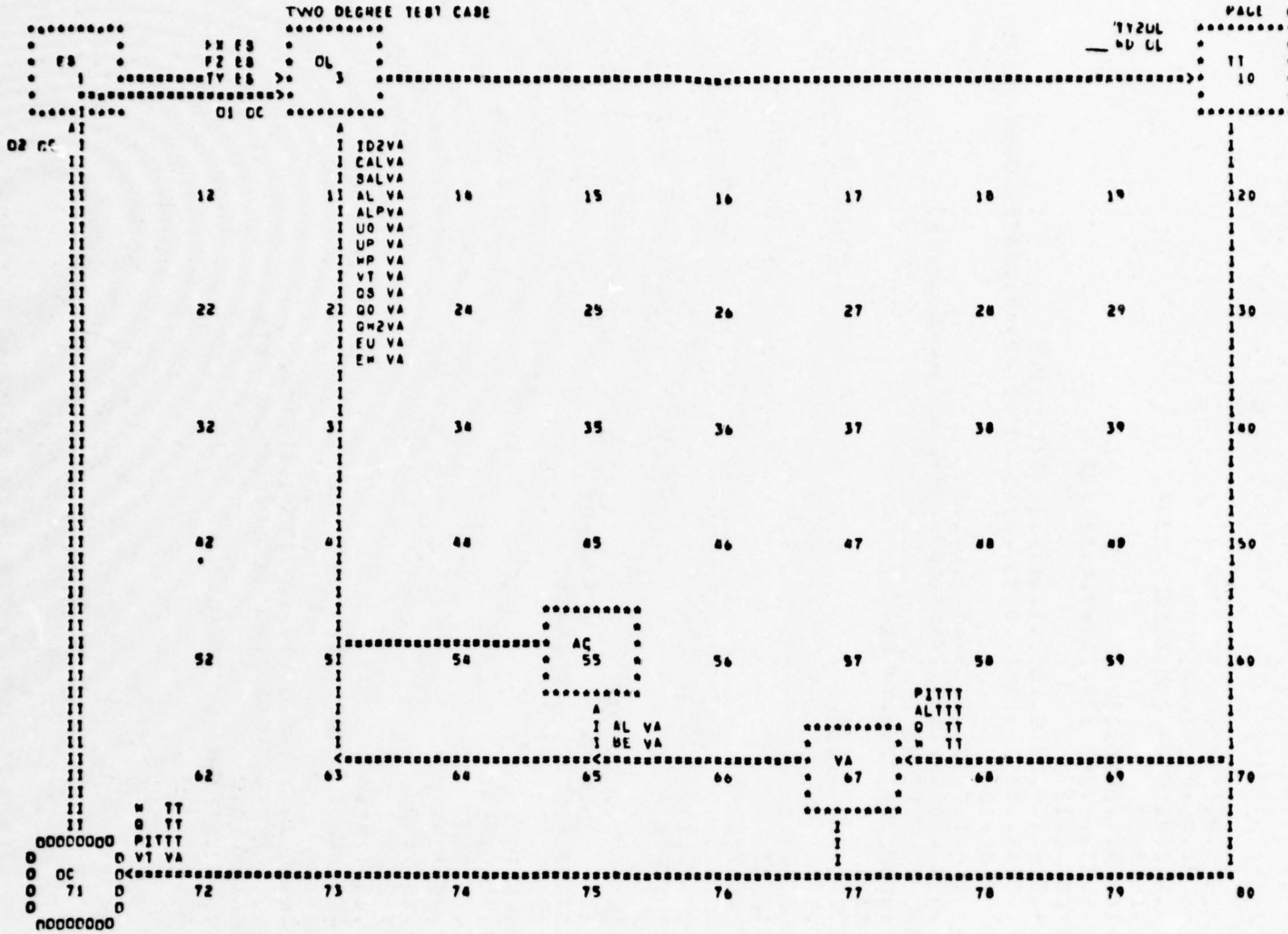


Figure 22 Jindivik 2DOF Airplane and Trim Components

```
PARAMETER VALUES
IDGVA=2,IBLES=1,IFNES=1,ISWOL=2
ID1VA=3,VS VA=726,ALSVA=0,S  VA=76,U  VA=720.41,ROLVA=0
TCOES=1,GAXES=1,GAZES=0,XO ES=-10.9,ZO ES=-.375,FX1ES=0
P2 ES=0
IYYTT=1811,ROLTT=0,U  TT=720.41
C  OL=4,MA1OL=83.92,XP1OL=0
MDECL=.642,MQ OL=-.246,ZDEOL=.2594
TABLE,CLTAC,7
-15,-10,-5,0,5,10,15
-1.27,-.758,-.243,.272,.787,1.3,1.82
TABLE,CDTAC,12
-1.0,-.5,-.1,-.05,0,.05,.1,.15,.2,.5,1.0,1.5
.0775,.0565,.0397,.0382,.0373,.0368,.0368,.0368,.0375,.0488,.0675,.0863
TABLE,CMTAC,10
-1.0,-.5,-.1,0,.1,.2,.3,.5,1.0,1.5
.134,.094,.0615,.0535,.0465,.04,.034,.022,-.008,-.038
INITIAL CONDITIONS
ALTTT=30000,PITTT=1,TH ES=377
O.C. DATA
YOP=30000,720.4,1,0,0
UOP=0,377
Q=1,100,1,1,1
RU=.01,.01
PRINT CONTROL=3
DESIGN O.C.
LINEAR ANALYSIS
STEADY STATE
XIC-X
PRINTER PLOTS,PLOT ON
PLOT ID=     M.K.WAHI      M/S 47-03
PLOT TABLES=CLTAC,CDTAC,CMTAC
DISPLAY1
AL VA,VS,TIME
W  TT,VS,TIME
U  TT,VS,TIME
Q  TT,VS,TIME
PITTT,VS,TIME
DISPLAY2
ALTTT,VS,TIME
TINC=.1,TMAX=12,PRATE=10
NO STATES
INT CONTROL=W  TT=1,Q  TT=1,PITTT=1,ALTTT=1
TITLE     FREE LONGITUDINAL MODEL,NONDIM. DATA
LINEAR ANALYSIS
INITIAL CONDITIONS=W  TT=60
SIMULATE
```

Figure 23 Jindivik 2DOF Longitudinal Model Data Inputs

/*/*/*/ STEADY STATE ANALYSIS /*/*/*/

A MAXIMUM OF 30 ITERATIONS CAN BE USED

TIME = 0.

STATES

1 X1 OC	= -21.888	2 X2 OC	= 1349.8	3 X3 OC	= -137.92	4 X4 OC	= -7.7845	5 X5 OC	= 481.05
6 TH ES	= 361.84	7 W TT	= -16.163	8 Q TT	= -.41426E-22	9 PITTT	= -1.2852	10 ALTTT	= 30492.

RATES

1 R1	= .63362E-09	2 R2	= -.21723E-04	3 R3	= -.32131E-07	4 R4	= -.69831E-08	5 R5	= .61234E-07
6 R6	= .18190E-11	7 R7	= -.12142E-09	8 R8	= -.46419E-08	9 R9	= -.41426E-22	10 R10	= .68212E-12

VARIABLES

1 UO VA	= 720.41	2 VO VA	= 0.	3 WO VA	= -16.163	4 PO VA	= 0.	5 QO VA	= -.41426E-22
6 RO VA	= 0.	7 ID2VA	= 3.0000	8 QW2VA	= 0.	9 RW2VA	= 0.	10 CALVA	= 1.0000
11 SALVA	= 0.	12 AL VA	= -1.2852	13 ALPVA	= -1.2852	14 VT VA	= 720.59	15 BE VA	= 0.
16 WP VA	= -16.163	17 UP VA	= 0.	18 EU VA	= 0.	19 EV VA	= 0.	20 EW VA	= 32.166
21 SIGVA	= .36761	22 QC VA	= 258.32	23 QS VA	= 17240.	24 MACVA	= .72584	25 ZO AC	= -.13962
26 XO AC	= -.36800E-01	27 MO AC	= .43925E-01	28 YB AC	= 2.0000	29 LB AC	= 2.0000	30 NB AC	= 2.0000
31 O1 OC	= -3.7445	32 O2 OC	= 361.84	33 FX ES	= 361.84	34 FZ ES	= 0.	35 TY ES	= -135.69
36 FSPES	= 0.	37 FSTES	= 0.	38 PPUES	= 0.	39 TPUES	= 0.	40 W2 ES	= 0.
41 T2 ES	= 0.	42 FX2OL	= -272.59	43 FZ2OL	= -2699.4	44 TY2OL	= -.14672E-06	45 UD OL	= -3.2483
46 WD OL	= -.12142E-09	47 MA2OL	= 83.920	48 XP2OL	= 0.	49 QD TT	= -.46419E-08		

PARAMETERS

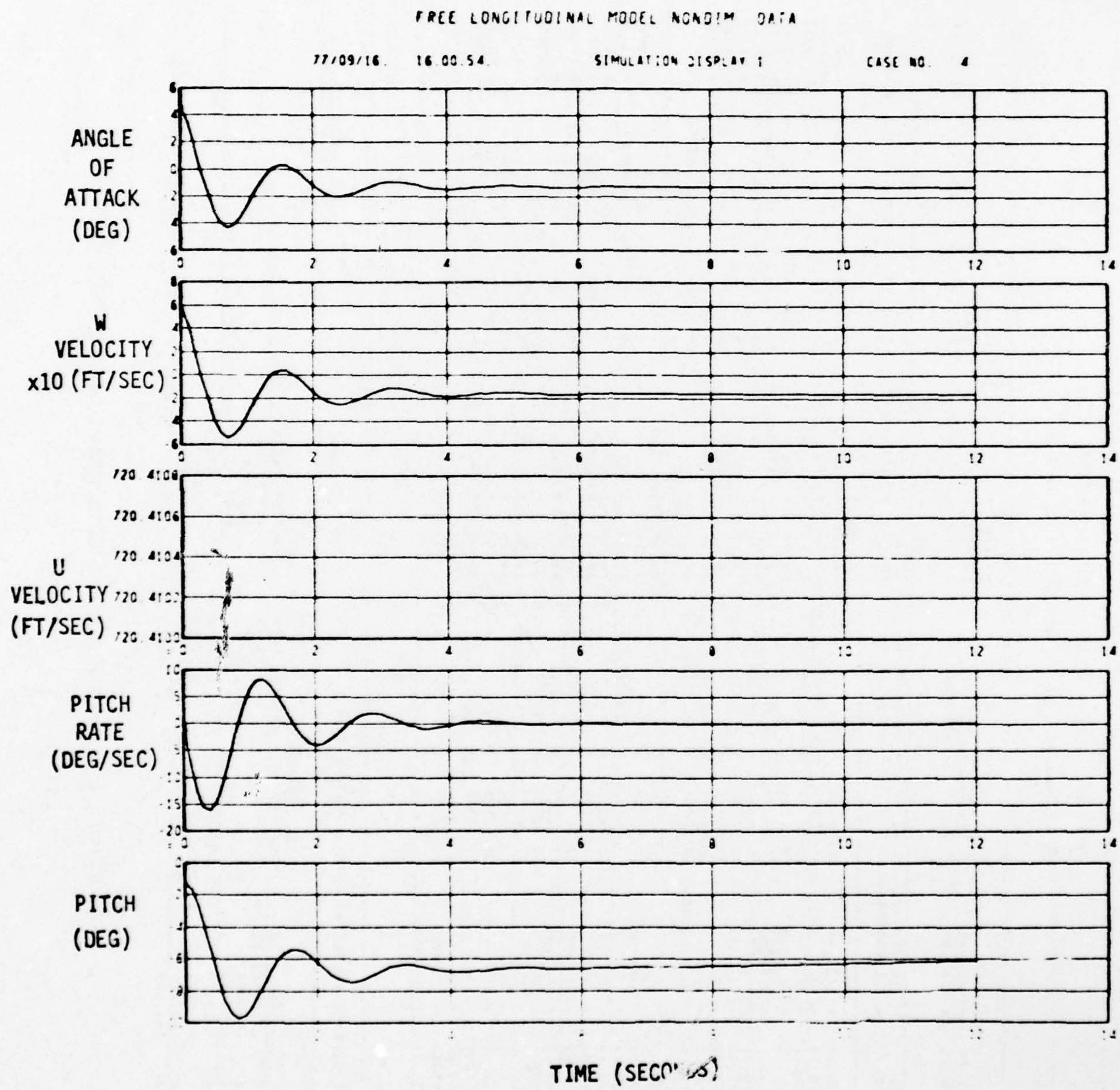
1 U VA	= 720.41	2 V VA	= 0.	3 P VA	= 0.	4 R VA	= 0.	5 ROLVA	= 0.
6 ID1VA	= 3.0000	7 VS VA	= 726.00	8 ALSVA	= 0.	9 S VA	= 76.000	10 UW VA	= 0.
11 VW VA	= 0.	12 WW VA	= 0.	13 PW VA	= 0.	14 QW1VA	= 0.	15 RW1VA	= 0.
16 IDGVA	= 2.0000	17 TCOES	= 1.0000	18 AMNES	= 0.	19 GAKES	= 1.0000	20 GAZES	= 0.
21 XO ES	= -10.900	22 ZO ES	= -.37500	23 PAMES	= 14.700	24 TAMES	= 459.00	25 PZ ES	= 0.
26 IFNES	= 1.0000	27 IBLES	= 1.0000	28 FX1ES	= 0.	29 XA OL	= 0.	30 XU OL	= 0.
31 XDEOL	= 0.	32 XTROL	= 0.	33 XSPOL	= 0.	34 XGEOL	= 0.	35 KXBOL	= 1.0000
36 ZA OL	= 0.	37 ZADOL	= 0.	38 ZQ OL	= 0.	39 ZU OL	= 0.	40 ZDEOL	= .25940
41 ZTROL	= 0.	42 ZSPOL	= 0.	43 ZGEOL	= 0.	44 KZBOL	= 1.0000	45 ZDSOL	= 0.
46 MALOL	= 0.	47 MADOL	= 0.	48 MQ OL	= -.24600	49 MU OL	= 0.	50 MDEOL	= .64200
51 MTROL	= 0.	52 MSPOL	= 0.	53 MGEOL	= 0.	54 KMBOL	= 1.0000	55 MDSOL	= 0.
56 MB OL	= 0.	57 KGEOL	= 0.	58 MA1OL	= 83.920	59 C OL	= 4.0000	60 XP1OL	= 0.
61 ISWOL	= 2.0000	62 STAOL	= 0.	63 SPOOL	= 0.	64 IYYTT	= 1811.0	65 ROLTT	= 0.
66 U TT	= 720.41								

SYSTEM EIGENVALUES AT THIS OPERATING POINT

10 EIGENVALUES

	REAL	IMAGINARY	NATURAL FREQ.	DAMPING RATIO
1	-.195558E-01	+- .176897	.177975	.109879
2	-.999990	0.	.999990	1.00000
3	-1.00003	0.	1.00003	1.00000
4	-14.0329	+- 5.68346	15.1402	.926867
5	-77.0371	+- 218.542	231.723	.332454
6	-140.624	0.	140.624	1.00000
7	-299.485	0.	299.485	1.00000

Figure 24 Trim Conditions Determined by the O.C. (2DOF)



Fi... Jindivik Free L...gitudinal Time History Plots (2DOF)

```
MODEL DESCRIPTION     THREE DEGREE LATERAL TEST CASE
LOCATION=67      VA   INPUTS=FD
LOCATION=55      AC   INPUTS=VA
LOCATION=37      DL   INPUTS=VA,AC
LOCATION=10      FD   INPUTS=DL
END OF MODEL
PRINT
```

Figure 26 Four DOF Model for 3DOF Lateral Simulation

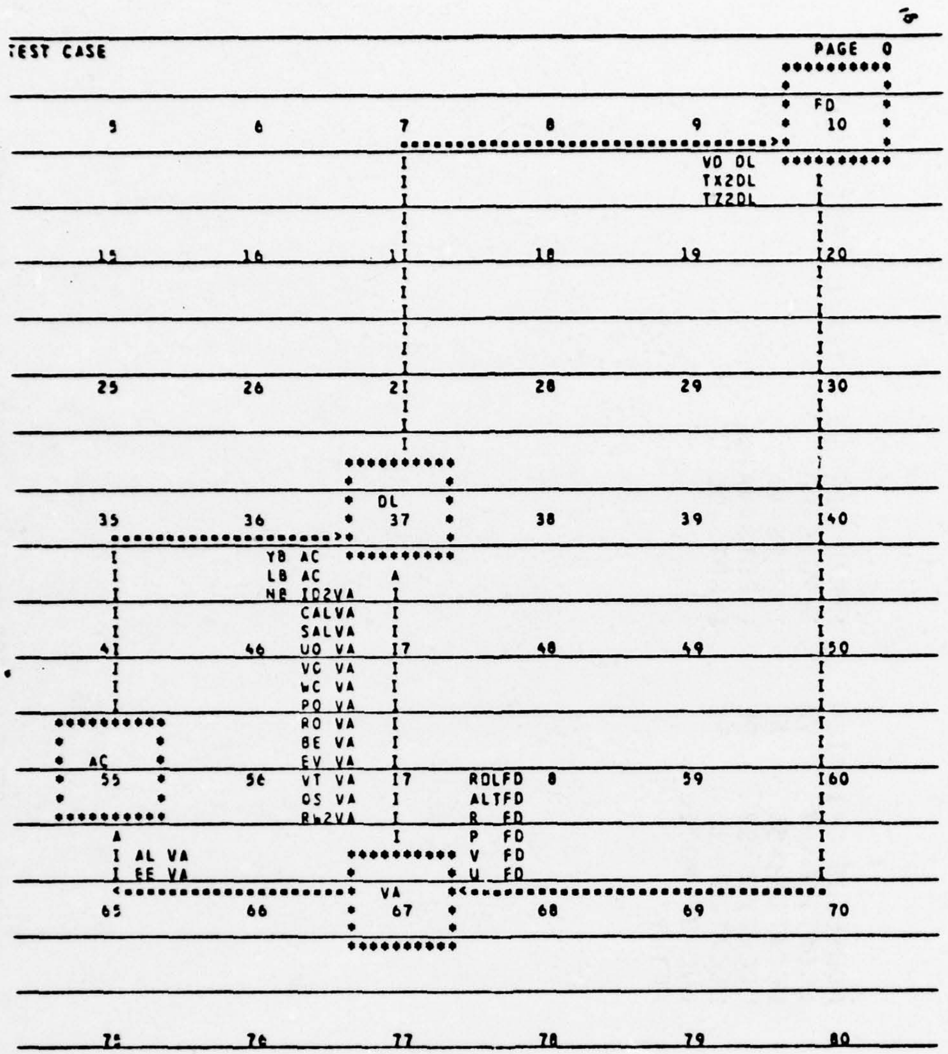


Figure 27 Schematic of 4DOF Model Usage

```
PARAMETER VALUES
MA CL=83.92
YP DL=-.0215,LP DL=-.370,NP DL=.00167,LR DL=.4498,NR DL=-.1545
LDACL=-.14184,NCADL=.0152
IDGVA=4
ID1VA=3,VS VA=726,ALSVA=0,S  VA=76
IXXFD=1190,IZZFD=2840,IXZFD=-200
B  CL=19
UD FD=0,W  VA=-16.43,PITVA=-1.3062,PITFD=-1.3062
TABLE,CYTAC,7
-90,-75,-60,0,60,75,90
-.75,-.75,-.75,-.75,-.75,-.75,-.75
TABLE,CNBAC,3
-20,0,20
.125,.08,.035
TABLE,CLBAC,3
-20,0,20
.188,-.072,-.332
INITIAL CONDITIONS
U  FD=720.41,ALTFD=30027
PRINT CONTROL=3
PRINTER PLOTS,PLOT ON
PLOT ID=     M.K.WAHI      MAIL STOP  47-03
PLOT TABLES=CYTAC,CNBAC,CLBAC
DISPLAY1
BE VA,VS,TIME
R  FD,VS,TIME
P  FD,VS,TIME
ROLFD,VS,TIME
ALTFD,VS,TIME
TINC=.1,TMAX=12,PRATE=10
NO STATES
INT CONTROL=V  FD=1,P  FD=1,R  FD=1,ROLFD=1
TITLE    FREE LATERAL MODEL
LINEAR ANALYSIS
INITIAL CONDITION=V  FD=16.3
SIMULATE
```

Figure 28 4DOF Model Data Input

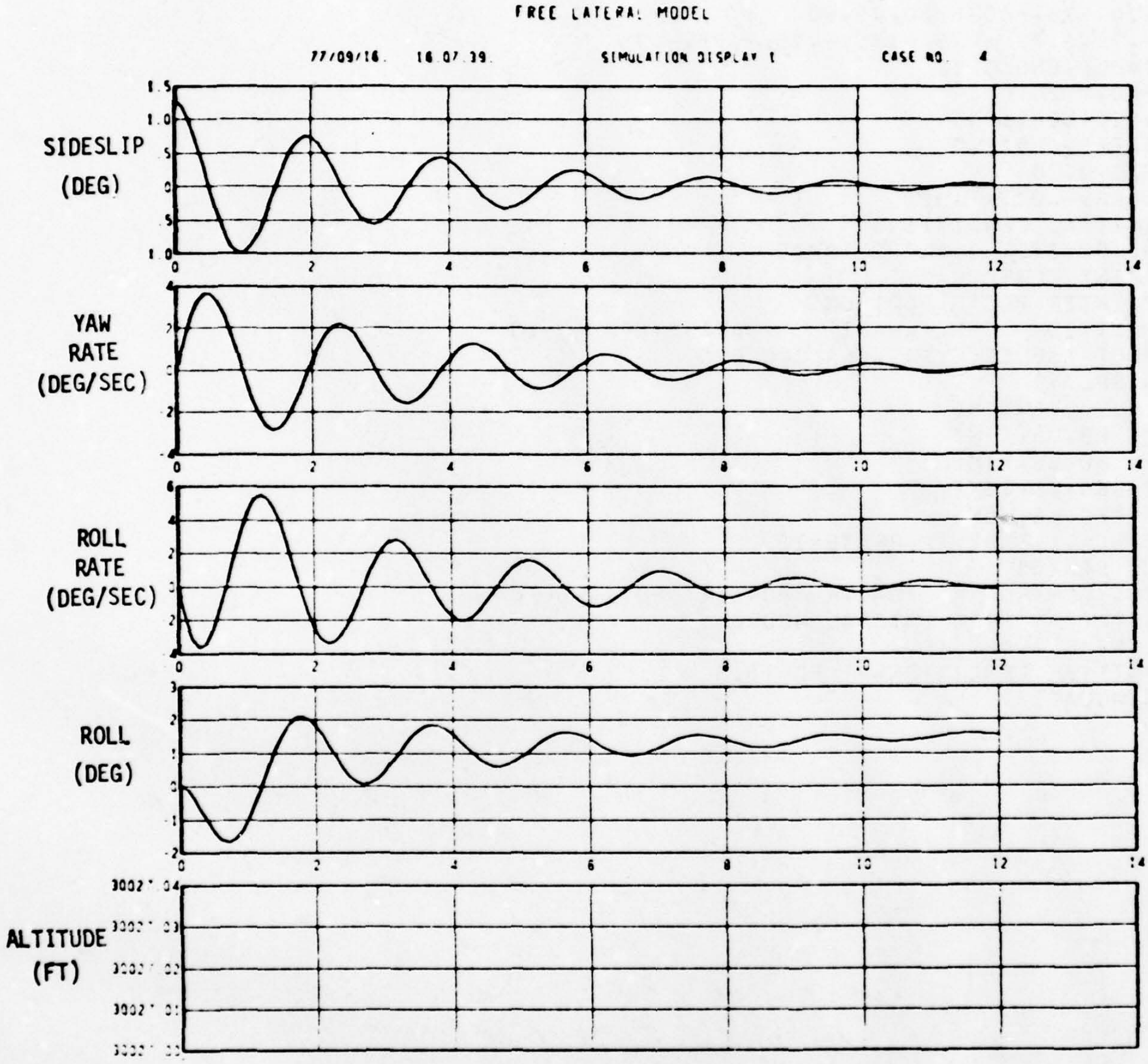


Figure 29 4DOF Model Time History Plots

```
MODEL DESCRIPTICN      THREE DEGREE LATERAL TEST CASE
LOCATION=67     VA     INPUTS=TD
LOCATION=55     AC     INPUTS=VA
LOCATION=37     DL     INPUTS=VA,AC
LOCATION=10     TD     INPUTS=DL
END OF MODEL
PRINT
```

Figure 30 Jindivik 3DOF Lateral Model

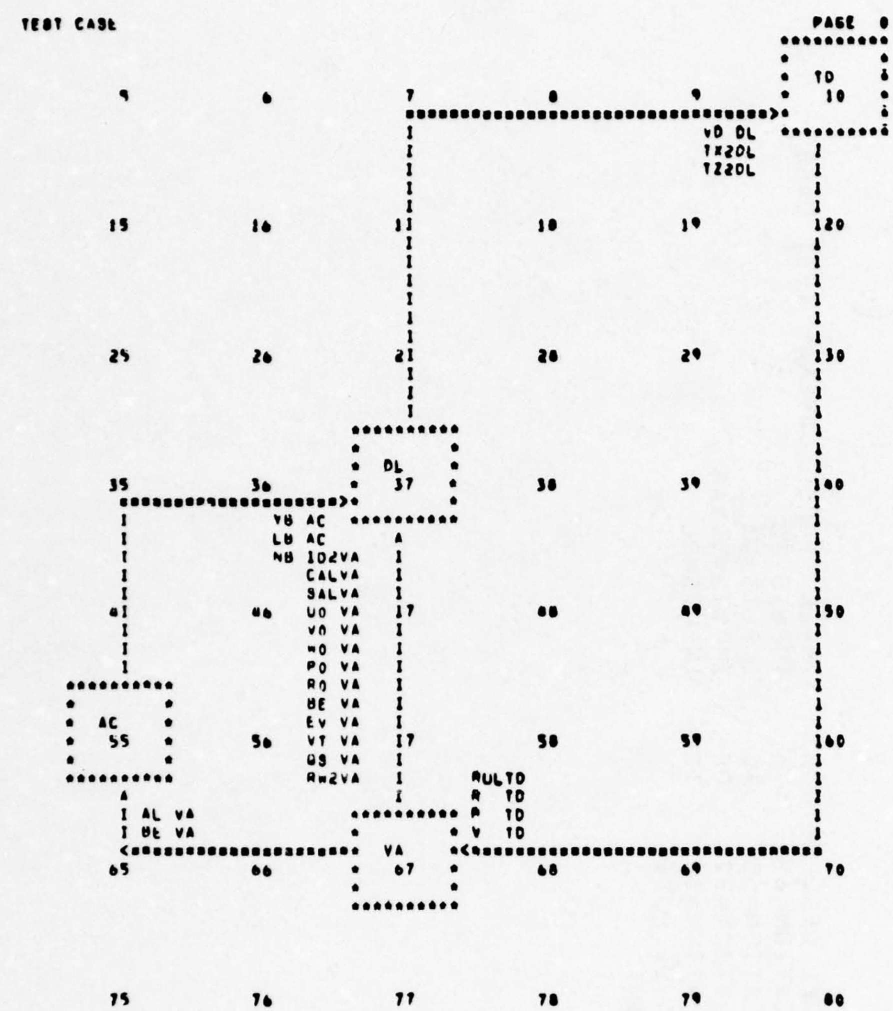


Figure 31 Schematic of Jindivik 3DOF Lateral Model

```
PARAMETER VALUES
MA DL=83.92
YP DL=-.0215,LP DL=-.370,NP DL=.00167,LR DL=.4498,NR DL=-.1545
LDADL=-.14184,NDADL=.0152
IDGVA=5
ID1VA=3,VS VA=726,ALSVA=C,S  VA=76
IXXTD=1190,IZZTD=2840,IXZTD=-200
B  DL=19
U  VA=720.41,W  VA=-16.43,ALTVA=30C27,PITVA=-1.3062,PITTD=-1.3062
TABLE,CYTAC,7
-90,-75,-60,0,60,75,90
-.75,-.75,-.75,-.75,-.75,-.75,-.75
TABLE,CNBAC,3
-20,0,20
.125,.08,.035
TABLE,CLBAC,3
-20,0,20
.188,-.072,-.332
INITIAL CONDITICNS
V  TD=0,P  TD=0,R  TD=0,ROLTD=0,YAWTD=0
PRINT CCNTROL=3
PRINTER PLOTS,PLOT ON
PLOT ID=    M.K.WAHI     MAIL STOP  47-03
PLOT TABLES=CYTAC,CNBAC,CLBAC
DISPLAY1
BE VA,VS,TIME
R  TD,VS,TIME
P  TD,VS,TIME
ROLTD,VS,TIME
ALTTC,VS,TIME
TINC=.1,TMAX=12,PRATE=10
NO STATES
INT CONTROL=V  TD=1,P  TD=1,R  TD=1,ROLTD=1
TITLE    FREE LATERAL MODEL
LINEAR ANALYSIS
INITIAL CONDITION=V  TD=16.3
SIMULATE
```

Figure 32 Model Data Input for 3DOF Lateral Model

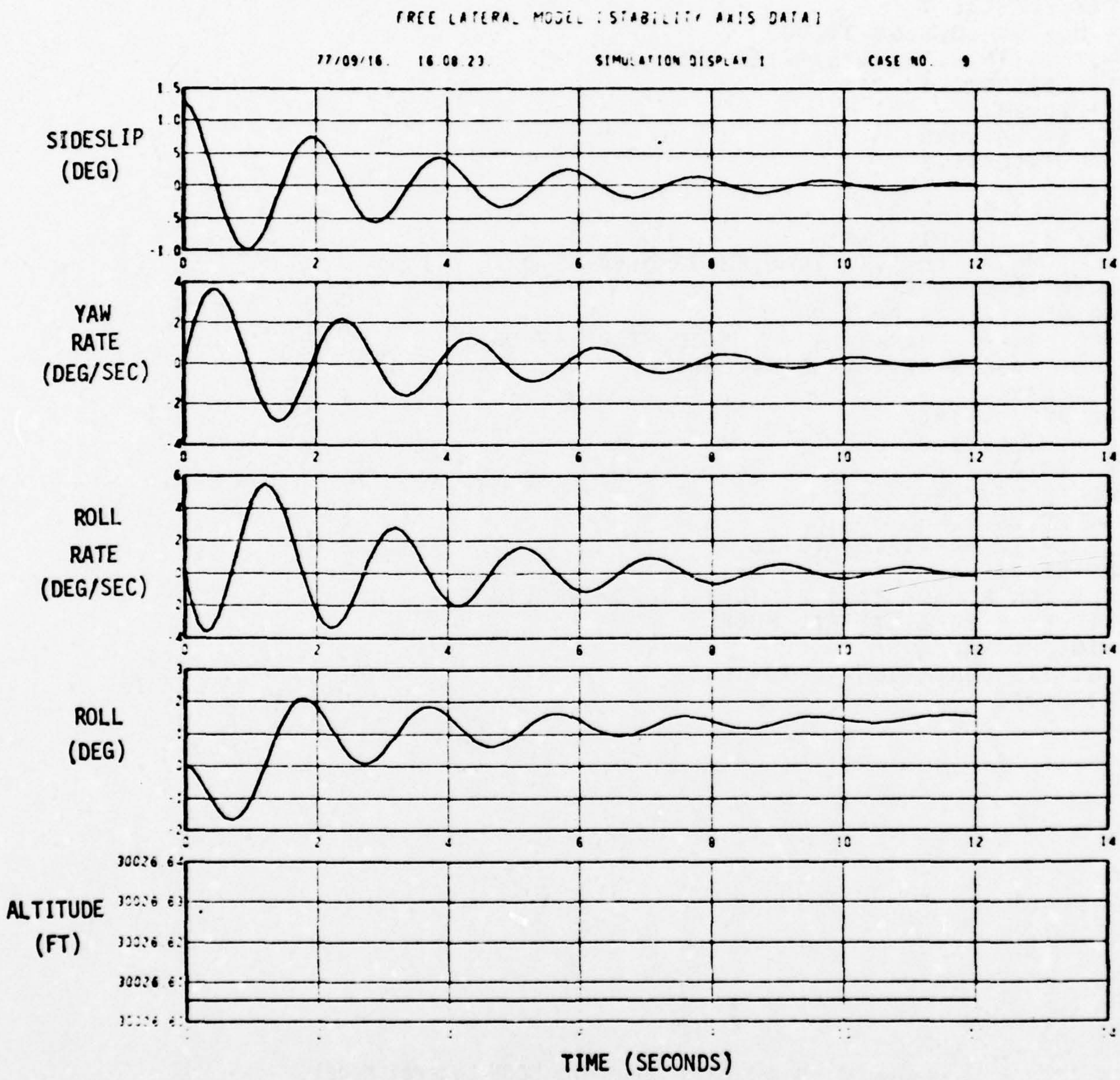


Figure 33 Time History Plots for 3DOF Lateral Model

SECTION III

JINDIVIK DROP TEST SIMULATION

3.1 Objectives

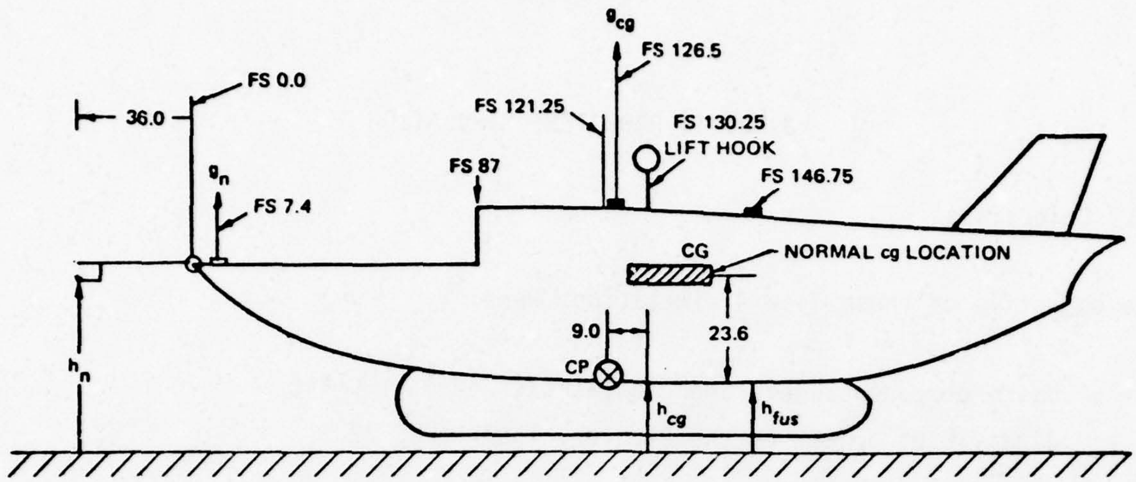
The objective of these Task 4 simulations were:

- o Using components developed in Tasks 1 and 3 simulate Jindivik Drop Test on the Recovery Trunk ACRS #2
- o Compare AFFDL Test (GFP) and EASY Simulation results and achieve $\pm 10\%$ correlation

3.2 Background/Technical Approach

Jindivik drop test data in the form of strip charts were received from AFFDL. The data for the six drop tests provided (Test No. 22, 27, 28, 29, 30 and 32) were reviewed. Drop Test No. 27 was selected as the most appropriate for the correlation purposes. The Jindivik drop test configuration and instrument locations are shown in Figure 34 and was extracted from Reference 4. The drop test data associated with test no. 27 are as follows:

Aircraft weight = 2,614 lbs.
Moments of Inertia
I_{xx} = 1190 slug ft^2
I_{yy} = 1810 slug ft^2
I_{zz} = 2840 slug ft^2
I_{xz} = -200 slug ft^2
Height of fuselage above ground at cg (h_{cg}) before drop is 30.5 inches
cg is under lift hook
Attitude
Pitch = 0 deg
Roll = 0 deg



The instrument pickups for g_w and h_w were placed on the starboard wi , tip opposite FS 141 and FS 131, respectively. Both were positioned 164.75 inches outboard of the aircraft centerline. For configurations 10, 11, and 12; h_w was moved to a position 156 inches outboard of the centerline.

Figure 34 Instrument Locations on Test Aircraft

Bleed air flow rate (assumed) = 0.80 lb/sec

Cushion vent area (geometric) = 53.3 sq. in.

Coefficient of discharge for vent = 0.6

Ambient air pressure = 14.7 psia

Discharge coefficient for flow through gap between trunk and ground = 0.9

Discharge coefficient for free portion of trunk perforations = 0.6

Discharge coefficient for flattened portion of trunk perforations = 0.2

Discharge coefficient for relief valve = 0.9

Body station of trunk axis = 157.5 inches

Water line of trunk axis = 0 inches

Body Station of c.g. = 120.25 inches

Water line of c.g. = 33.6 inches

The accelerometer traces for pitch and heave (drop #27) are shown in Figure 35. These data were digitized and integrated to obtain displacements h_n and h_{cg} as shown on Figure 36. The differences between recorded and integrated data are obvious. Also note that the -1g level is never reached during free fall; the average level reached was -.85g from .05 to .25 seconds (see Figure 35) at which time the ground contact occurs. In addition, the trunk air supply pressure, temperature and flow rate were not recorded during the drop test. In spite of these discrepancies a genuine effort was made to obtain the best correlation possible with the available drop test data and are explained in the following paragraphs. The trunk and cushion pressures recorded for the drop test #27 are shown in Figures 37. Figure 38 shows the trunk relief valve characteristics and Figure 39 shows the air supply system and the cushion vent arrangement.

3.3 Simulation Description and Results

The basic modules required to model the airplane and the trunk cushion dynamics for a drop test are components SG and TK (no aerodynamic effects). SG contains the rigid body dynamics for integrating the aircraft states and is driven by the body axis accelerations and torques resulting from the trunk,

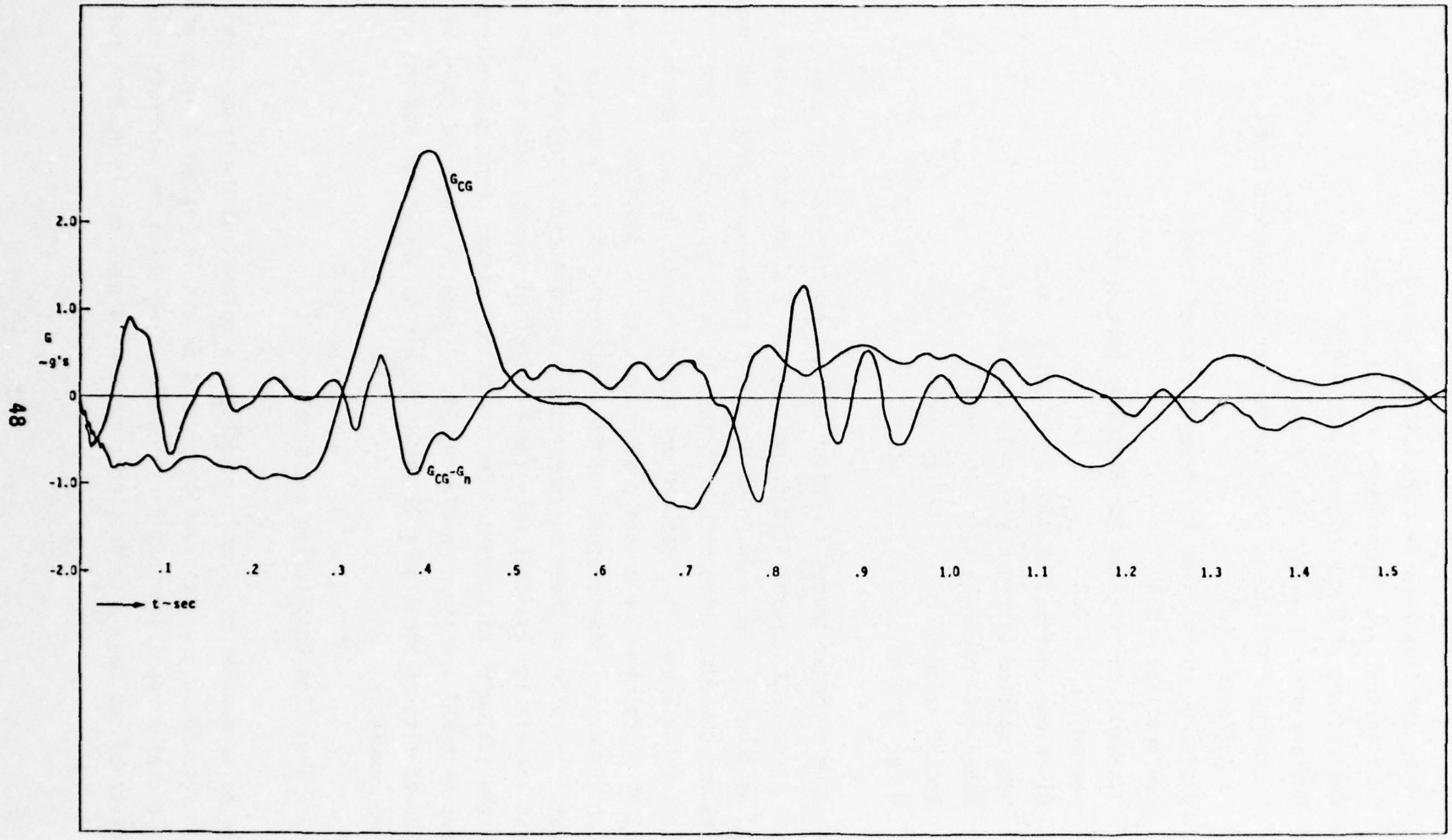


Figure 35 Accelerometer Traces for Pitch and Heave

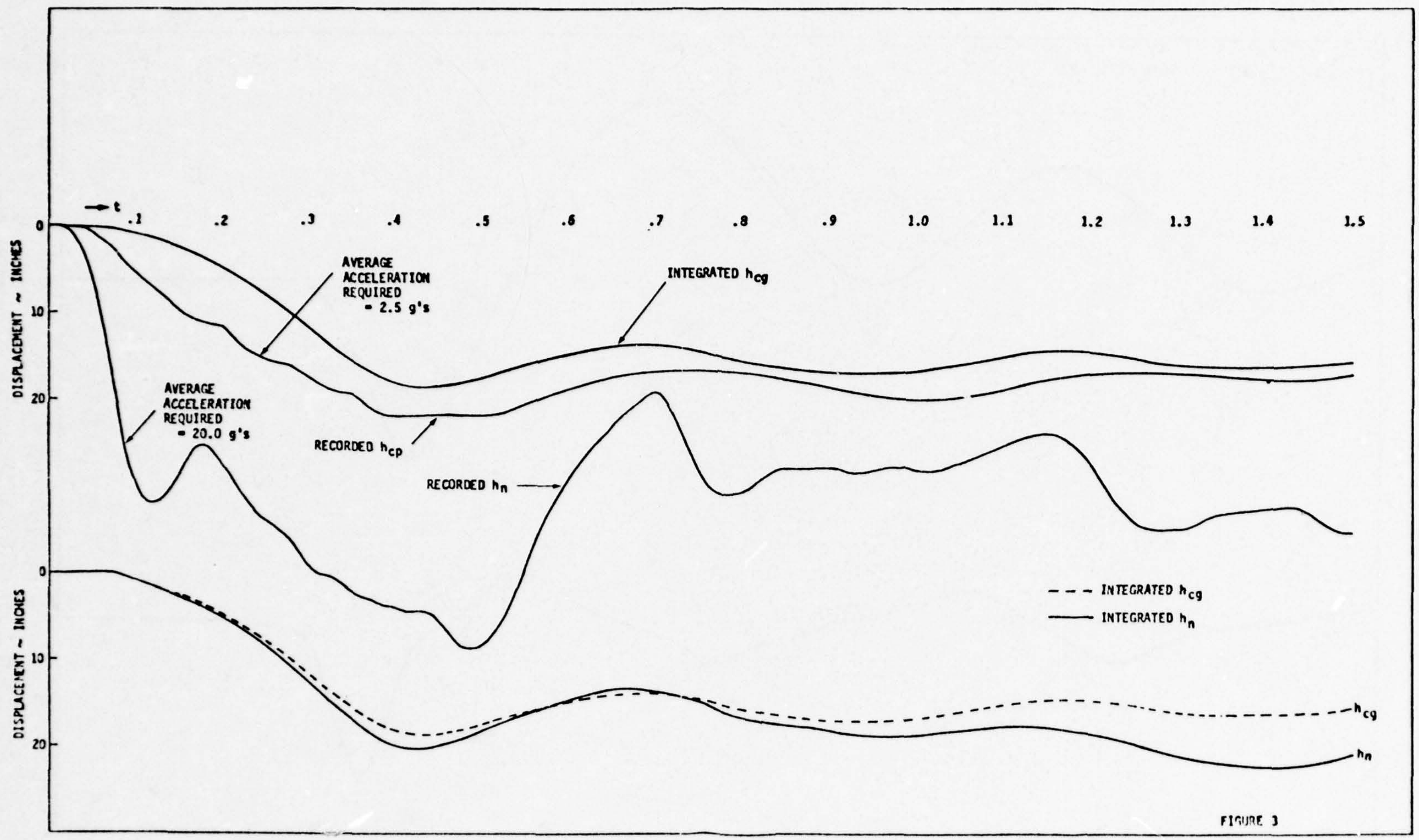


Figure 36 Integrated vs Recorded Data for Drop No. 27

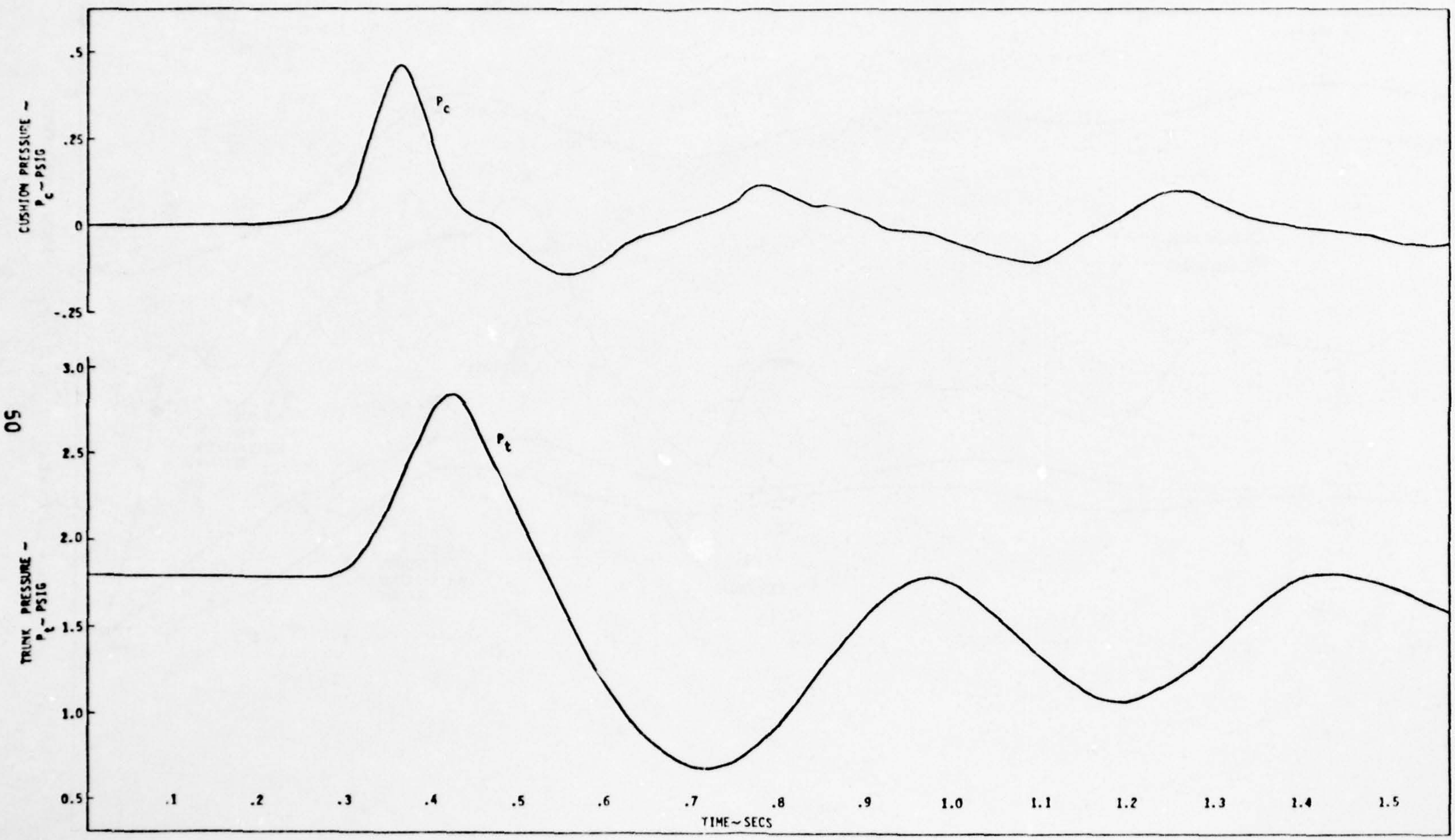


Figure 37 Recorded Pressure Data for Drop No, 27

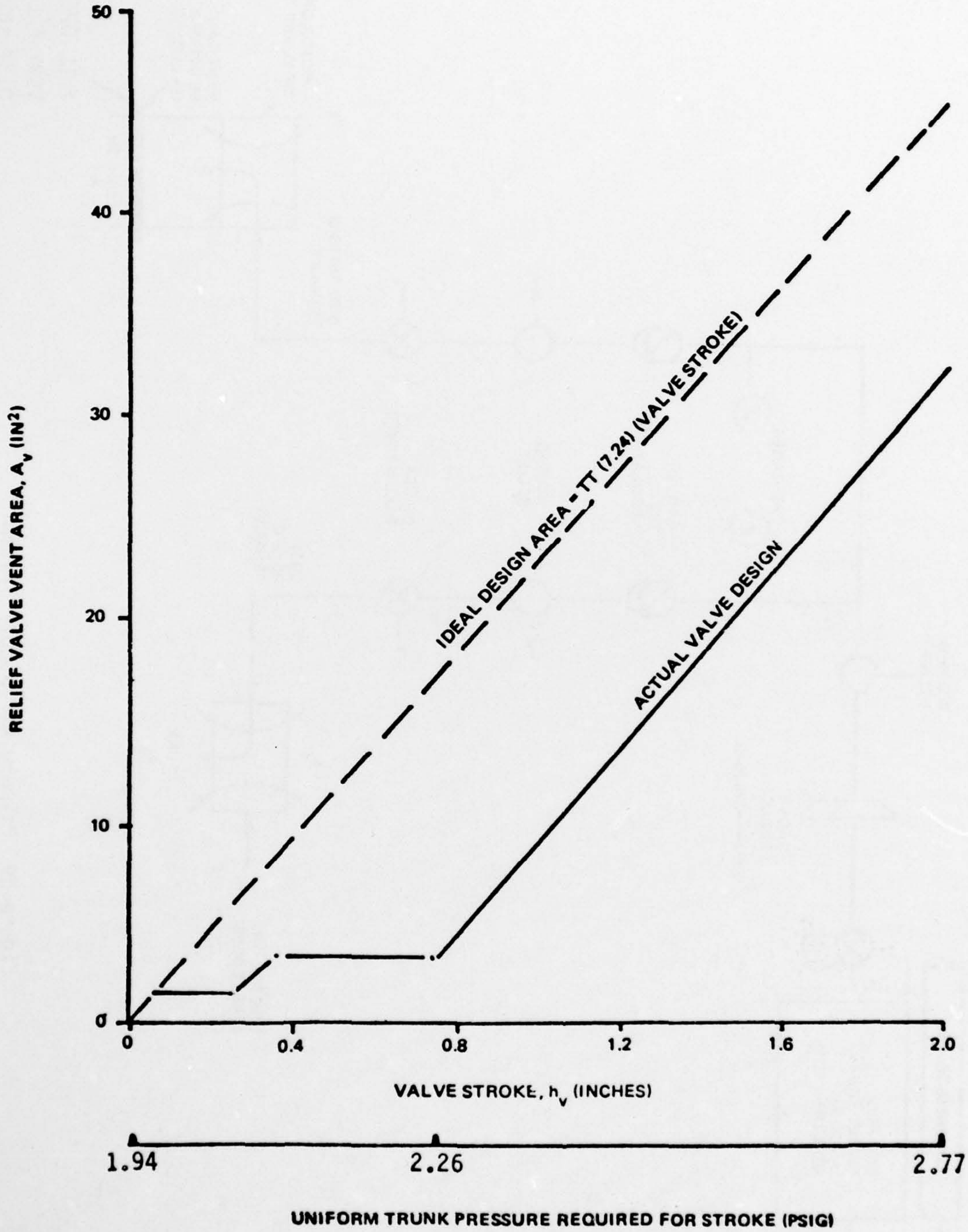
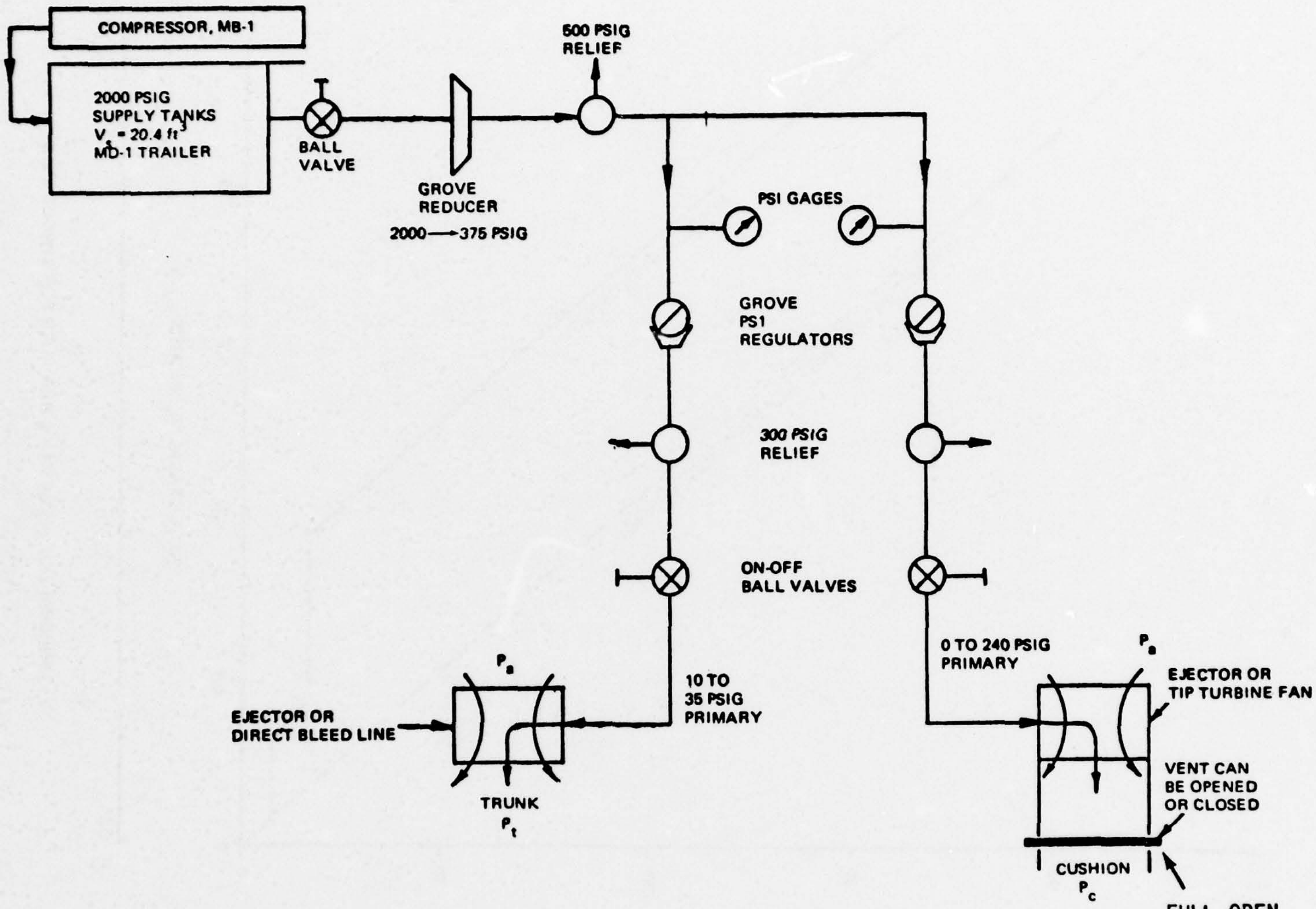


Figure 38 Relief Valve Vent Area Versus Stroke



FULL OPEN
VENT AREA=
0.37 sq. ft.
or 53.3 sq. in.

Figure 39 Primary Air Supply System

the angular rates and the gravity force. The total accelerations, normally calculated in aerodynamic components OL and DL, are calculated, in this model, by using Fortran statements.

Figure 40 shows the model description for this drop test simulation. The ADD TABLES command was used to simulate the free fall acceleration from .05 to .25 seconds. GTAB is the name of the table and -35 signifies a one dimensional table with a maximum of 16 pairs of data points allowed. The ADD VARIABLES = GT command is necessary since the acceleration GT is a variable and a table lookup function of time. The ADD PARAMETERS command allows inputs for vehicle mass and the cushion vent area. The vehicle mass is used to convert the trunk forces into accelerations. The cushion vent area is used to calculate the flow rate from cushion to atmosphere in the CALL FNFLOW command. FNFLOW is a standard subroutine for calculating flow. The output WCUTK is automatically connected to the trunk module TK. The module TK also needs as inputs U, V, W, P, Q, R, ROL, PIT, YAW, X and ALT made available by SG. The next set of Fortran Statements is for calculating the three accelerations and are automatically connected to module SG. The three torque outputs from trunk TK are directly supplied to SG. This completes the model definition. Figure 41 shows the EASY generated model diagram.

The input data used to simulate this model is shown in Figure 42. Following the command PARAMETER VALUES are the input data values for the components TK and SG. The free fall acceleration table GTAB and various table arrays for the trunk TK follow. The following is the procedure used to arrive at the data for these tabular arrays. Figure 43 shows a top view schematic of the ACRS #2 trunk used in the drop test. For this simulation the trunk was divided into two identical halves. Each half was then divided into twenty segments or elements. These elements were grouped into 4 sets each signifying a group of elements of same basic shape but different dimensions so that the scaling concept could be applied, see Figure 44. Using these concepts and the coordinate system shown in Figure 45 the relevant information is extracted from Figure 46 using the trunk parameter definitions shown in Figures 47 and 48. Table 2 lists the Jindivik trunk parameter data input into model TK. The non-zero INITIAL CONDITIONS for states complete the model definition.

```
MODEL DESCRIPTION        ACRS NO. 2 DROP TEST
ADD TABLES=GTAB,-35
ADD VARIABLES=GT
ADD PARAMETERS=AMASS,CAC
FORTRAN STATEMENTS
      CALL FNFLOW(PA TK,PC TK,TCUTK,CAC,1.,FN,WCUTK)
LOCATION=65     TK      INPUTS=SG
FORTRAN STATEMENTS
      GT=TBLU1(TIME,GTAB(4),GTAB(20),1,-16)
      UD SG=FXTTK/AMASS-(Q  SG*W  SG-R  SG*V  SG)*.01745-
     1                          32.2*SIN(PITSG*.01745)
      VD SG=FYTTK/AMASS-(R  SG*U  SG-P  SG*W  SG)*.01745
     1           +32.2*COS(PITSG*.01745)*SIN(ROLSG*.01745)
      WD SG=FZTTK/AMASS-(P  SG*V  SG-Q  SG*U  SG)*.01745
     1               +GT*COS(PITSG*.01745)*COS(ROLSG*.01745)
LOCATION=20     SG      INPUTS=TK(TXT=TX,TYT=TY,TZT=TZ)
END OF MODEL
PRINT
```

Figure 40 Jindivik Drop Test Model Definition

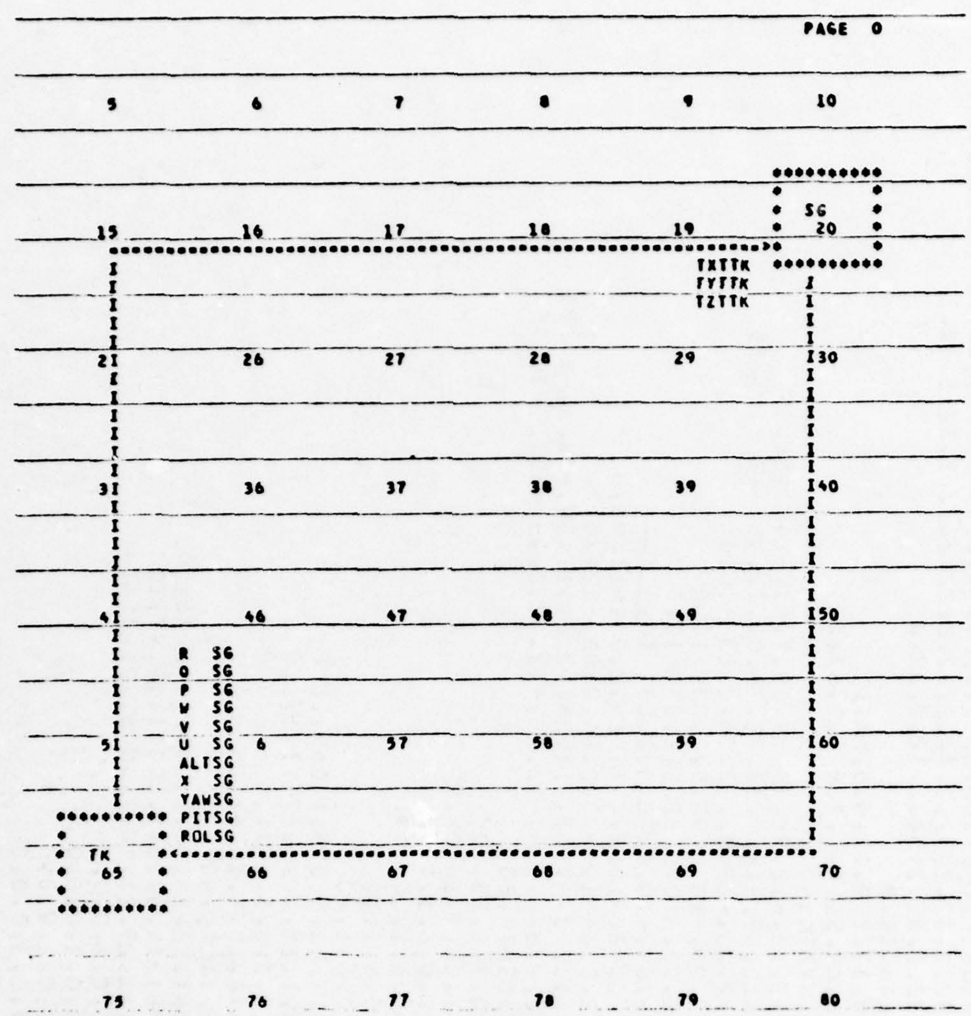


Figure 41 Jindivik Drop Test Model Schematic

```
PARAMETER VALUES
AMASS=81.26,AMOTK=C,DMPTK=.025
PA TK=14.7,CAC=32.C,TCUTK=560,WTRTK=48.
TTRTK=560,NE TK=-20,CDGTK=.9,NSTTK=4
NPTTK=10,BSTTK=157.5,WLTTK=0,CD1TK=.6
CD2TK=.2,CDATK=.9,BSCTK=130.25,WLCTK=33.6
TAUTK=.005,EPCTK=1.,VU TK=6
IXXSG=1190,IYYSG=1810,IZZSG=2840,IXZSG=-200
TABLE,GTAB,16
0,.04,.055,.C65,.C8,.095,.12,.14,.15,.175,.185,.205,.21,.225,.25,2.
32.2,32.2,32.2,32.2,32.2,32.2,32.2,32.2,32.2,32.2
32.2,32.2,32.2,32.2,32.2,32.2
TABLE,ABLTK,8
3.1,.8,37,1,9.33,5.6,55,1
9.44,6.C9,56.9,1,7.8,3.1,63,1
TABLE,XYZTK,40
86.72,2.83,-12,78.75,83.89,7.61,-14.7,56.25
79.83,10.98,-16.9,33.75,75.C8,12.95,-17.1,11.25
69.25,2.03,-10,0.,62.75,2.C3,-10,0
56.5,2.03,-10,0,50.5,2.03,-10,0
44.5,2.C3,-10,0,38.5,2.03,-10,0
32.5,2.03,-10,0,26.5,2.03,-10,0
20.56,2.03,-10.5,C,14.69,2.03,-11.08,0
8.81,2.03,-11.66,C,2.94,2.03,-12.24,0
-2.73,13.73,-19.7,-11.25,-7.61,11.39,-18.3,-33.75
-12.72,8.5,-15.3,-56.25,-12.46,2.48,-13.1,-78.75
TABLE,DSMTK,30
7,1,.0,7,1.23,.0
7,1.41,.0,7,1.42,.C
6.5,1,.0,6.5,1,.0
6,1,.0,6,1,.C
6,1,.0,6,1,.C
6,1,.0,6,1,.0
5.875,1,.0,5.875,1.055,.0
5.875,1.11,.C,5.875,1.17,.0
8.8,1,.0,8.3,.93,.C
7.7,.78,.0,7.1,.66,.0
TABLE,IALTK,40
1,.0061,12.88,10,1,.0061,16.76,10
1,.0061,19.8,10,1,.0061,19.97,10
2,.0061,29.21,10,2,.0061,29.21,10
2,0,C,C,2,0,0,0
2,0,0,0,2,0,0,0
2,0,0,0,2,0,0,C
3,0,0,0,3,0,0,0
3,0,0,C,3,0,0,0
4,0,0,C,4,0,C,0
4,0,0,C,4,0,0,0
TABLE,RELTK,2
1.94,2.77
0.,32.4
INITIAL CONDITIONS
PITSG=0,ALTSG=4.5C8
PT TK=16.7,VT TK=46,PC TK=14.7,VC TK=35
ERROR CONTROLS
PT TK=.001,VT TK=.001,PC TK=.0C1,VC TK=.001
U  SG=.001,V  SG=.001,W  SG=.001
P  SG=.001,Q  SG=.C01,R  SG=.0C1
ROLSG=.0001,PITSG=.0001,YAWSG=.0001
X  SG=.001,ALTSG=.C001
```

Figure 42 Drop Test Simulation Input Data

```
NO STATES
INT CONTROL=PT TK=1,VT TK=1,VC TK=1,PC TK=1
STEADY STATE
XIC-X
INT CONTROL
W  SG=1,PITSG=1,ALTSG=1,Q  SG=1
PRINT CONTROL=4
LINEAR ANALYSIS
PRINTER PLOTS
PLOT ID=    M.K.WAHI      M/S 47-03
SI MANUAL SCALES
DISPLAY1
ALTSG,VS,TIME,XRANGE=0,1.2,YRANGE=1.6,5.6
PITSG,VS,TIME,XRANGE=0,1.2
R7,VS,TIME,XRANGE=C,1.2,YRANGE=-140,60
FZTTK,VS,TIME,XRANGE=0,1.2
TYTTK,VS,TIME,XRANGE=0,1.2
DISPLAY2
PT TK,VS,TIME,XRANGE=0,1.2,YRANGE=14,19
PC TK,VS,TIME,XRANGE=0,1.2,YRANGE=14,19
VT TK,VS,TIME,XRANGE=0,1.2
VC TK,VS,TIME,XRANGE=0,1.2
WTATK,VS,TIME,XRANGE=0,1.2
DISPLAY3
WCATK,VS,TIME,XRANGE=0,1.2
WTCTK,VS,TIME,XRANGE=0,1.2
CPTTK,VS,TIME,XRANCE=0,1.2
TINC=.01,TMAX=1.2,PRATE=5,INT MODE=5
SIMULATE
XIC-X
LINEAR ANALYSIS
```

Figure 42 Drop Test Simulation Input Data (Concluded)

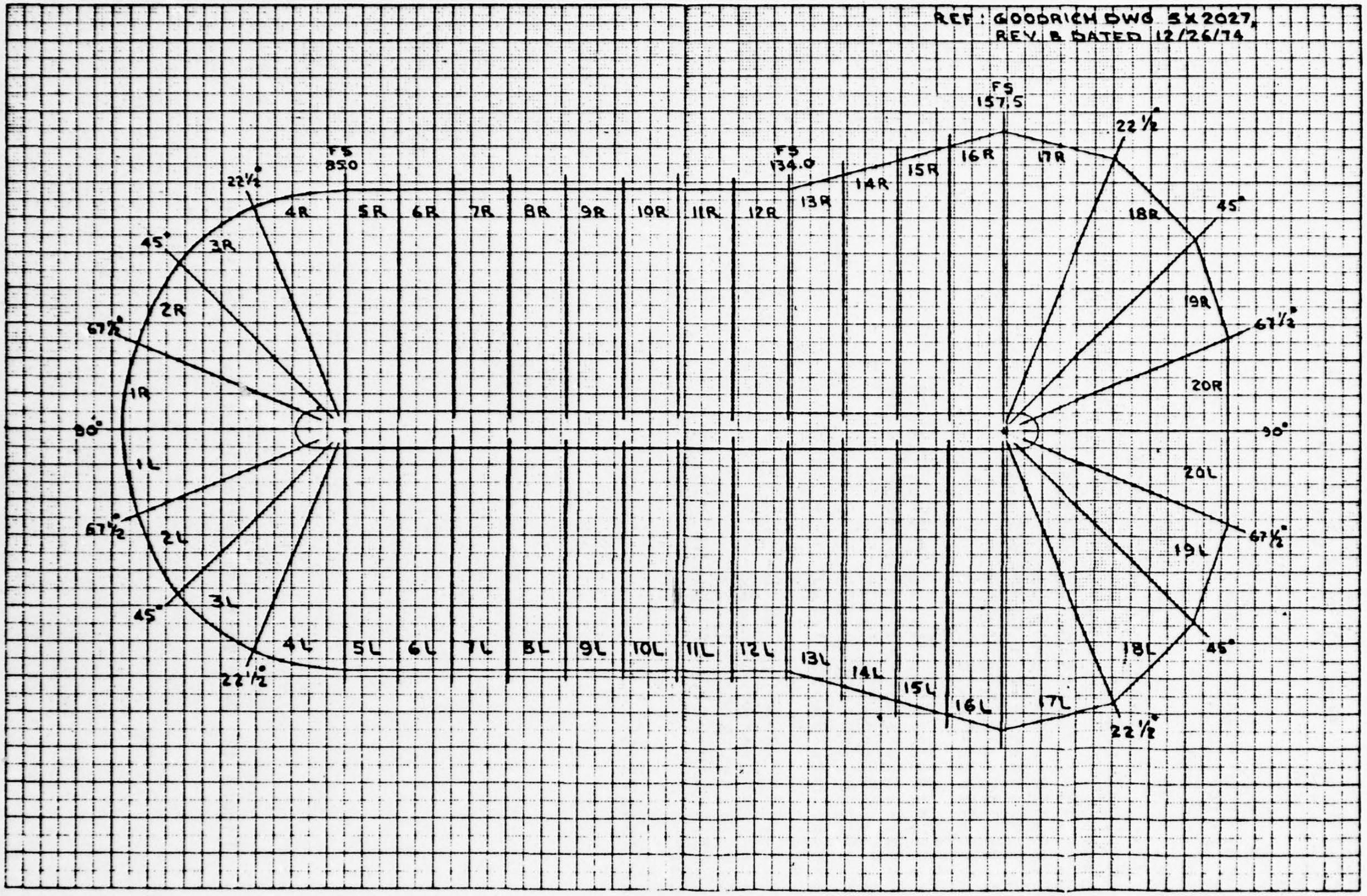
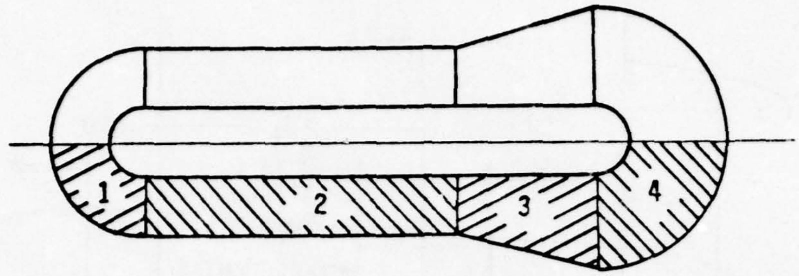
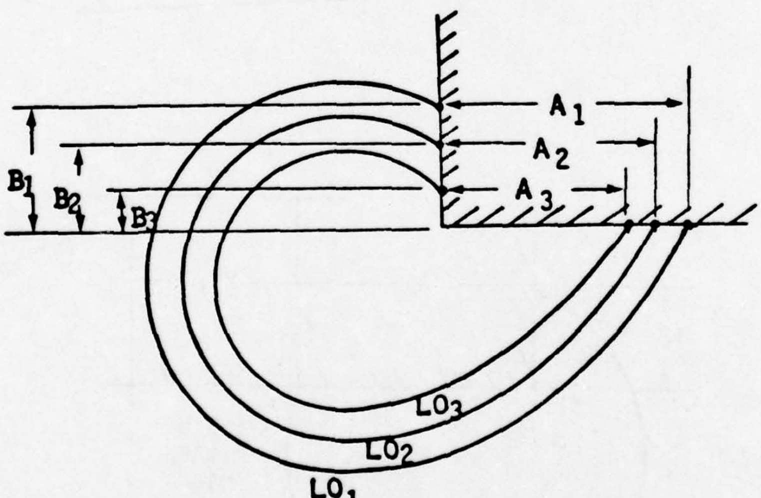


Figure 43 Plan View of ACRS No. 2



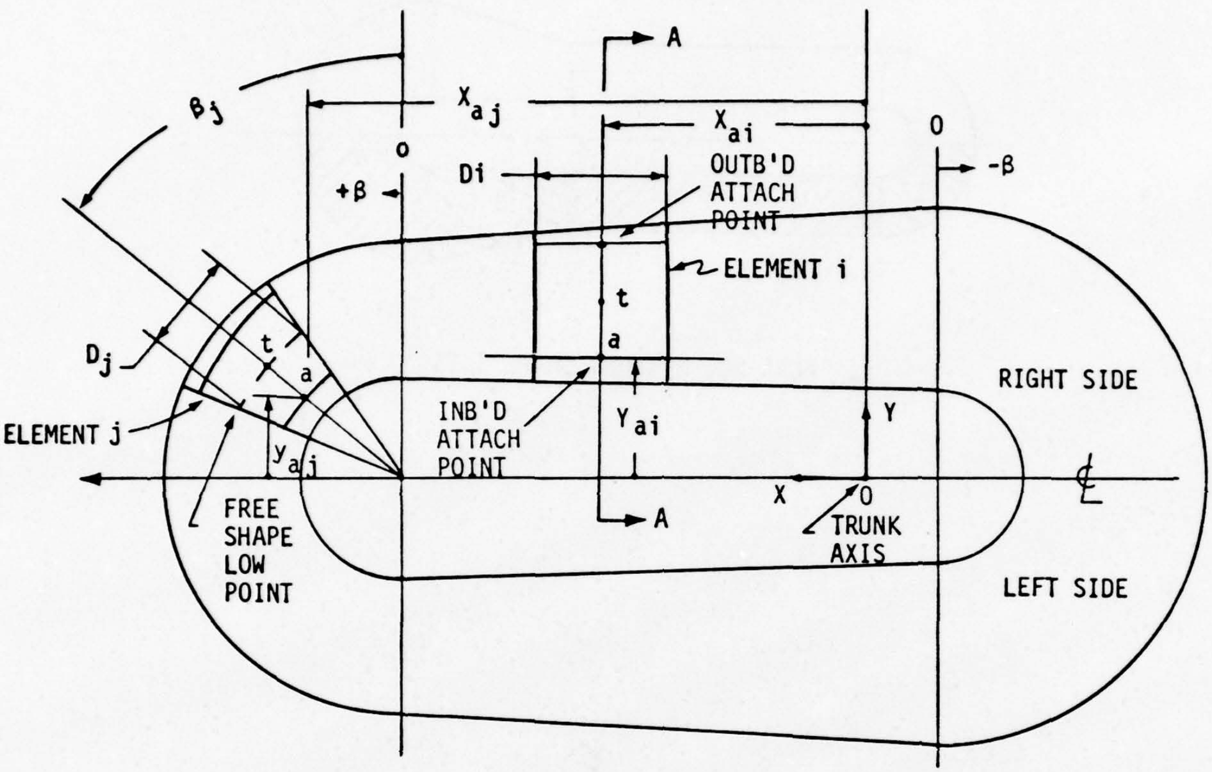
a. ACRS NO. 2 TRUNK SHAPE SETS

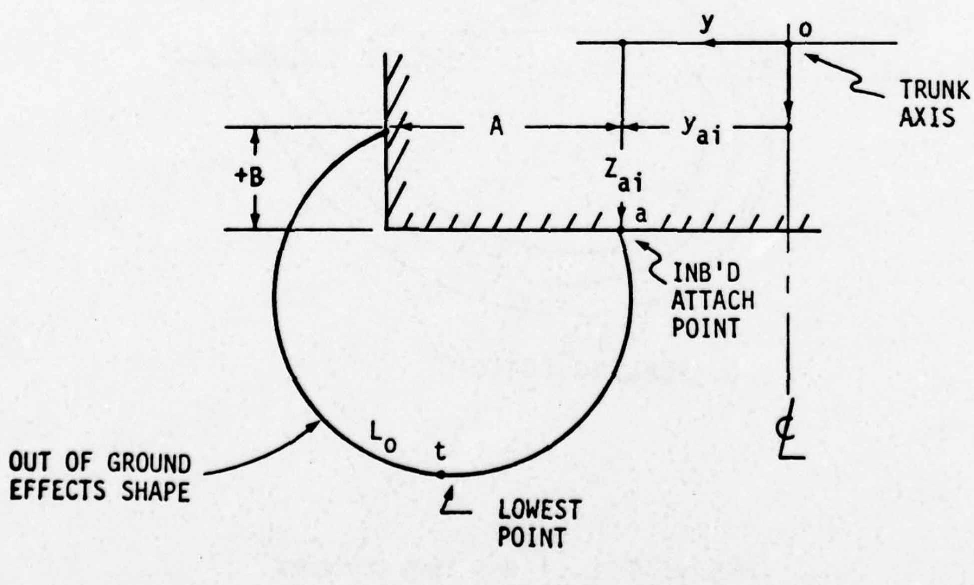


b. SCALING CONCEPT

Figure 44 Trunk Modeling Concepts

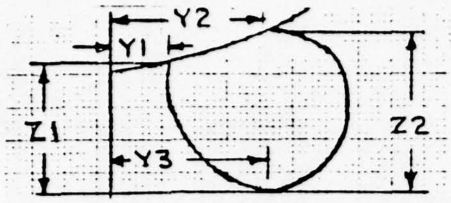
TOP VIEW





SECTION A - A

Figure 45 Trunk Element Locations and Coordinate System



REF: GOODRICH DWG 5X2027
REV. B DATED 12/26/74
ACRS NO. 2
TREAD NOT INCLUDED

ZONE	LOCATION	Y1	Z1	Y2	Z2	Y3	LENGTH
1	FS85.0,90° FS85.0,67 1/2°	5.1	10.3	17.6	12.8	17.6	37.7
2	FS85.0,67 1/2° FS85.0,45°	3.6	10.2	15.6	15.5	17.6	45.4
3	FS85.0,45° FS85.0,22 1/2°	2.4	10.1	13.2	16.9	17.6	49.9
4	FS85.0,22 1/2° FS85.0,0°	2.1	10.0	12.0	16.3	17.6	54.3
5	FS85.0 FS91.125	2.03	10.00	11.36	15.60	17.6	55.0
6	FS91.125 FS97.25	2.03	10.00	11.36	15.60	17.6	55.0
7	FS97.25 FS103.375	2.03	10.00	11.36	15.60	17.6	55.0
8	FS103.375 FS109.5	2.03	10.00	11.36	15.60	17.6	55.0
9	FS109.5 FS115.625	2.03	10.00	11.36	15.60	17.6	55.0
10	FS115.625 FS121.75	2.03	10.00	11.36	15.60	17.6	55.0
11	FS121.75 FS127.875	2.03	10.00	11.36	15.60	17.6	55.0
12	FS127.875 FS134	2.03	10.00	11.36	15.60	17.6	55.0
13	FS134 FS139.9	2.03	10.50	11.47	16.59	18.1	56.9
14	FS139.9 FS145.8	2.03	11.00	11.68	17.29	19.1	60.7
15	FS145.8 FS151.6	2.03	11.56	11.89	17.99	20.1	68.1
16	FS151.6 FS157	2.03	11.8	12.1	18.69	21.1	70.4
17	FS157.5,0° FS157.5,22 1/2°	2.08	12.0	13.4	19.1	21.8	69.2
18	FS157.5,22 1/2° FS147.5, 45°	2.4	12.1	14.0	18.6	21.0	64.4
19	FS157.5, 45° FS157.5,67 1/2°	2.7	12.3	16.3	17.2	19.4	53.2
20	FS157.5,67 1/2° FS157.5, 90°	4.0	12.4	17.9	15.1	17.9	44.1

Figure 46 Trunk Geometry Information, ACRS No. 2

- A, B, LO DEFINE PROPERTIES FOR EACH SET

ACRS NO. 2

SET NO.	DIMENSIONS A	INCHES B	LO	DEFINES ELEMENTS FROM-TO	I_{TYPE} (1)
1	3.10	0.80	37.00	1-4	0 OR 1
2	9.33	5.60	55.00	5-12	1
3	9.44	6.09	56.90	13-16	1
4	7.80	3.10	63.00	17-20	0 OR 1

(1) I_{TYPE} = 0 FROZEN MODEL

= 1 MEMBRANE MODEL

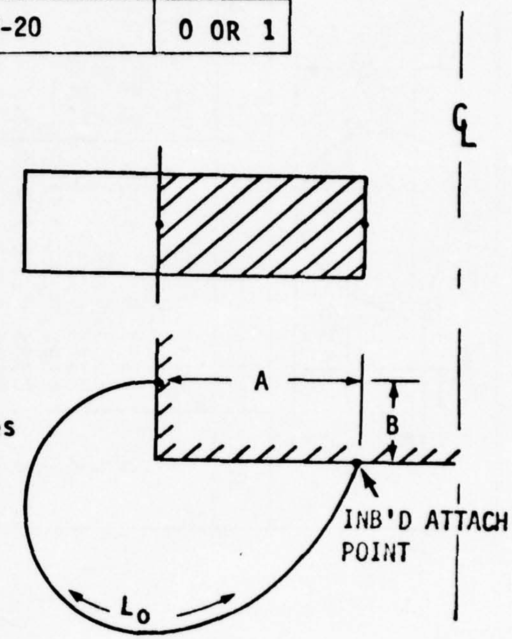


Figure 47 Definition of Trunk Element Section Properties

- ELEMENT PARAMETERS

  $X_a, Y_a, Z_a, \beta, D, I_s, S, A_p, L_p, L_h, \mu_x, \mu_y$

X_a, Y_a, Z_a	LOCATION OF INB'D ATTACH POINT (a)
β	ANGLE OFF LONGITUDINAL AXIS $\beta = 0$ SIDE $\beta \neq 0$ END
D	ELEMENT WIDTH
I_s	DATA SET NO. TO WHICH THE ELEMENT BELONGS
S	SCALING FACTOR (YO, ZO, L1, L3, ACV, AS)
A_p	ORIFICE AREA PER UNIT AREA OF PERFORATED SURFACE
L_p	PERIPHERAL DISTANCE FROM OUTB'D ATTACH POINT (b) TO EDGE OF PERFORATED AREA
L_h	WIDTH OF PERFORATED AREA
μ_x, μ_y	ELEMENT GROUND FRICTION COEFFICIENTS IN X, Y AXIS

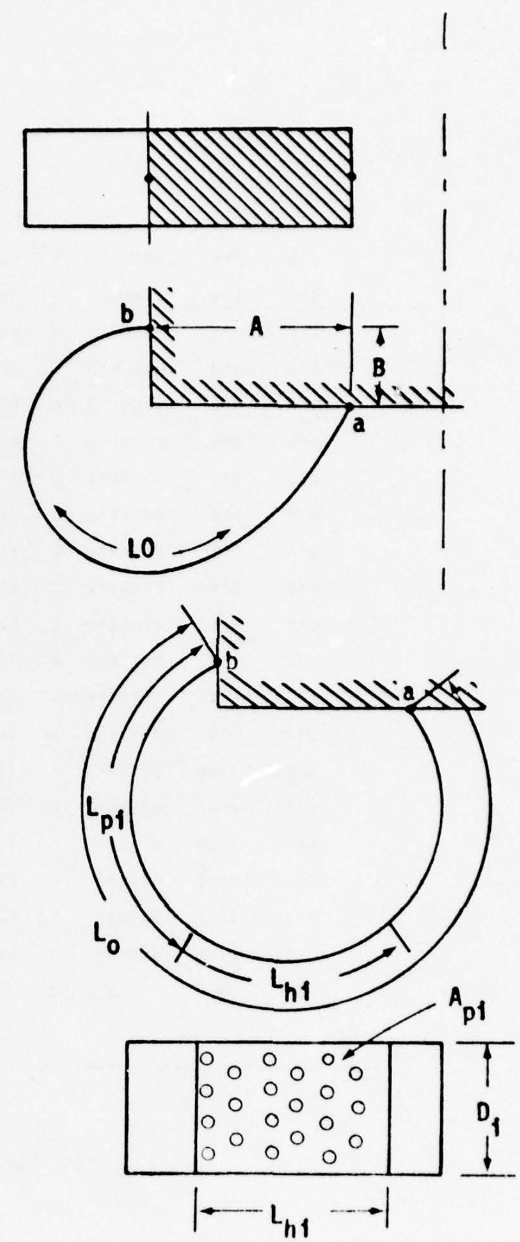


Figure 48 Trunk Element Specification

TABLE 2 TRUNK PARAMETER DATA ARRAYS, ACRS No. 2

••••• TRUNK PARAMETER DATA ••

ELEMENT	XA	YA	ZA	RA	BET	D	S	MU	IS	AP	LF	LH
1	86.72	2.63	-12.90	14.49	78.75	7.00	1.000	0.000	1	.00610	12.9	10.0
2	83.89	7.61	-14.70	13.69	56.25	7.00	1.230	0.000	1	.00610	16.8	10.0
3	79.83	10.96	-16.90	13.20	33.75	7.00	1.410	0.000	1	.00610	19.6	10.0
4	75.08	12.95	-17.10	13.20	11.25	7.00	1.420	0.000	1	.00610	20.0	10.0
5	69.25	2.03	-10.00	2.03	0.00	6.50	1.000	0.000	2	.00610	29.2	10.0
6	62.75	2.03	-10.00	2.03	0.00	6.50	1.000	0.000	2	.00610	29.2	10.0
7	56.50	2.03	-10.00	2.03	0.00	6.00	1.000	0.000	2	0.00000	0.0	0.0
8	50.50	2.03	-10.00	2.03	0.00	6.00	1.000	0.000	2	0.00000	0.0	0.0
9	44.50	2.03	-10.00	2.03	0.00	6.00	1.000	0.000	2	0.00000	0.0	0.0
10	38.50	2.03	-10.00	2.03	0.00	6.00	1.000	0.000	2	0.00000	0.0	0.0
11	32.50	2.03	-10.00	2.03	0.00	6.00	1.000	0.000	2	0.00000	0.0	0.0
12	26.50	2.03	-10.00	2.03	0.00	6.00	1.000	0.000	2	0.00000	0.0	0.0
13	20.56	2.03	-10.50	2.03	0.00	5.88	1.000	0.000	3	0.00000	0.0	0.0
14	14.69	2.03	-11.08	2.03	0.00	5.88	1.055	0.000	3	0.00000	0.0	0.0
15	8.81	2.03	-11.66	2.03	0.00	5.88	1.110	0.000	3	0.00000	0.0	0.0
16	2.94	2.03	-12.24	2.03	0.00	5.88	1.170	0.000	3	0.00000	0.0	0.0
17	-2.73	13.73	-19.70	14.00	-11.25	8.80	1.000	0.000	4	0.00000	0.0	0.0
18	-7.61	11.39	-18.30	13.70	-33.75	8.30	.930	0.000	4	0.00000	0.0	0.0
19	-12.72	8.50	-15.30	15.30	-56.25	7.70	.780	0.000	4	0.00000	0.0	0.0
20	-12.46	2.46	-13.10	12.70	-78.75	7.10	.660	0.000	4	0.00000	0.0	0.0

The commands beginning with NO STATES and ending with STEADY STATE calculate a steady state condition of the test vehicle just before its release from the test rig. The steady state results are shown in Figure 49. The appropriate pitch and heave states are then turned on and a LINEAR ANALYSIS is commanded and Figure 50 shows the results. The eigenvalues and the damping ratios indicate the system to be stable. The on-line plotter is then turned on and the desired time-history plots are requested. The command SIMULATE then causes the drop test simulation and the resulting correlations between simulation and test data are presented in Figures 51 through 59.

Figures 51, 52 and 53 show a comparison of the drop test #27 with the EASY simulation results for a trunk model with frozen trunk end elements. Configuration 1 (see Table 3) has a full open vent (effective area = 32 in.^{2}). Configuration 2 has a part open vent (effective area = 14 in.^{2}). The part open vent condition brings the simulated cushion pressure inline with the drop test cushion pressure.

Figures 54, 55 and 56 show a comparison of the drop test #27 data with the EASY simulation results for a trunk model with constrained membrane trunk end elements for configurations 3 and 5. The constrained membrane simulation shows a much better matchup of trunk plunge frequencies and heave accelerations. The part open vent condition has 18 in.^{2} effective area in Figures 54, 55, 56 and not 14 in.^{2} as in Figures 51, 52 and 53. Note also that a lower damping coefficient was used in the part open vent condition to bring the trunk pressure peak in line with the full open vent run.

Figures 57, 58 and 59 show a comparison of the EASY trunk component with frozen end elements vs. constrained end elements (configurations 3 and 4 respectively). Both configurations (see Table 3) have identical effective cushion vent area (14 in.^{2}) and damping coefficient (.025).

```
/*/*/*/   STEADY STATE ANALYSIS   /*/*/*/

A MAXIMUM OF  3. ITERATIONS CAN BE USED


     TIME =  0.                                          STATES
 1 PT TK   =  16.506     2 VT TK   =  46.016     3 PC TK   =  14.700     4 VC TK   =  34.897     5 U  SG   =  0.
 6 V  SG   =  0.         7 W  SG   =  0.         8 P  SG   =  0.         9 Q  SG   =  0.        10 R  SG   =  0.
11 ROLSG   =  0.        12 PITSG   =  0.        13 YAWSG   =  0.        14 X  SG   =  0.        15 ALTSG   =  4.5080

                                                 RATES
 1 R1      = -.41632E-04   2 R2    =  .90467E-06   3 R3    =  .43260E-04   4 R4    = -.36270E-04   5 R5    =  0.
 6 R6      =  0.           7 R7    =  32.200       8 R8    =  0.           9 R9    =  0.          10 R10   =  0.
11 R11     =  0.          12 R12   =  0.          13 R13   =  0.          14 R14   =  0.          15 R15   =  0.


SYSTEM EIGENVALUES AT THIS OPERATING POINT


                         4  EIGENVALUES
          REAL           IMAGINARY       NATURAL FREQ.     DAMPING RATIO
   1    -1.14377         0.                 1.14377          1.00000
   2    -179.252         0.                 179.252          1.00000
   3    -200.000         0.                 200.000          1.00000
   4    -21798.7         0.                 21798.7          1.00000




        9.01206      CPU SECONDS WERE REQUIRED FOR THE PREVIOUS ANALYSIS

COMMAND CARD ----->     X1C-X
```

Figure 49 Steady State Condition Before Drop Release

/*/*/*/ LINEAR ANALYSIS /*/*/*/

STATE NAME	OPERATING POINT	PERTURBATION SIZE	INTEGRATOR CONTROL
1 PT TK	16.506	.100E-01	1
2 VT TK	46.016	.100E-01	1
3 PC TK	14.700	.100E-01	1
4 VC TK	34.697	.100E-01	1
5 U SG	0.	.100E-01	0
6 V SG	0.	.100E-01	0
7 W SG	0.	.100E-01	1
8 P SG	0.	.100E-01	0
9 Q SG	0.	.100E-01	1
10 R SG	0.	.100E-01	0
11 ROLSG	0.	.100E-02	0
12 PITSG	0.	.100E-02	1
13 YAWSG	0.	.100E-02	0
14 X SG	0.	.100E-01	0
15 ALTSG	4.5080	.100E-02	1

RATES AT OPERATING POINT

1 PT TK = -.41632E-04	2 VT TK = .90467E-06	3 PC TK = .43260E-04	4 VC TK = -.36270E-04	5 U SG = 0.
6 V SG = 0.	7 W SG = 32.197	8 P SG = 0.	9 Q SG = .35314E-02	10 R SG = 0.
11 ROLSG = 0.	12 PITSG = 0.	13 YAWSG = 0.	14 X SG = 0.	15 ALTSG = 0.

1 ELEMENTS OF /RATIO/ DIFFER FROM 1 BY 10/. THESE ELEMENTS ARE PRECEDED BY AN * IN THE STABILITY MATRIX
RATIO(5, 7) = .500046

STABILITY MATRIX

	PT TK	VT TK	PC TK	VC TK	W SG	Q SG	PITSG	ALTSG
PT TK	-1.196	86.07	47.69	0.	0.	0.	0.	0.
VT TK	.4973E-02	-200.0	-110.6	0.	0.	0.	0.	0.
PC TK	.5214	0.	-.2178E+05	101.1	0.	0.	-15.91	-2007.
VC TK	-.1994	0.	4434.	-200.0	0.	0.	31.46	3969.
W SG	.5707E-07	0.	-35.32	0.	0.	0.	*-.4903E-05	0.
Q SG	-.9996E-07	0.	43.52	0.	0.	0.	0.	0.
PITSG	0.	0.	0.	0.	0.	1.000	0.	0.
ALTSG	0.	0.	0.	0.	-1.000	0.	0.	0.

8 EIGENVALUES

	REAL		IMAGINARY	NATURAL FREQ.	DAMPING RATIO
1	-.371813E-08	+-	.244567E-02	.244567E-02	.152029E-05
2	-.915363E-02	+-	.543311E-01	.550968E-01	.166137
3	-1.14377		0.	1.14377	1.00000
4	-179.233		0.	179.233	1.00000
5	-200.000		0.	200.000	1.00000
6	-21798.7		0.	21798.7	1.00000

Figure 50 Linear analysis Before Drop Release

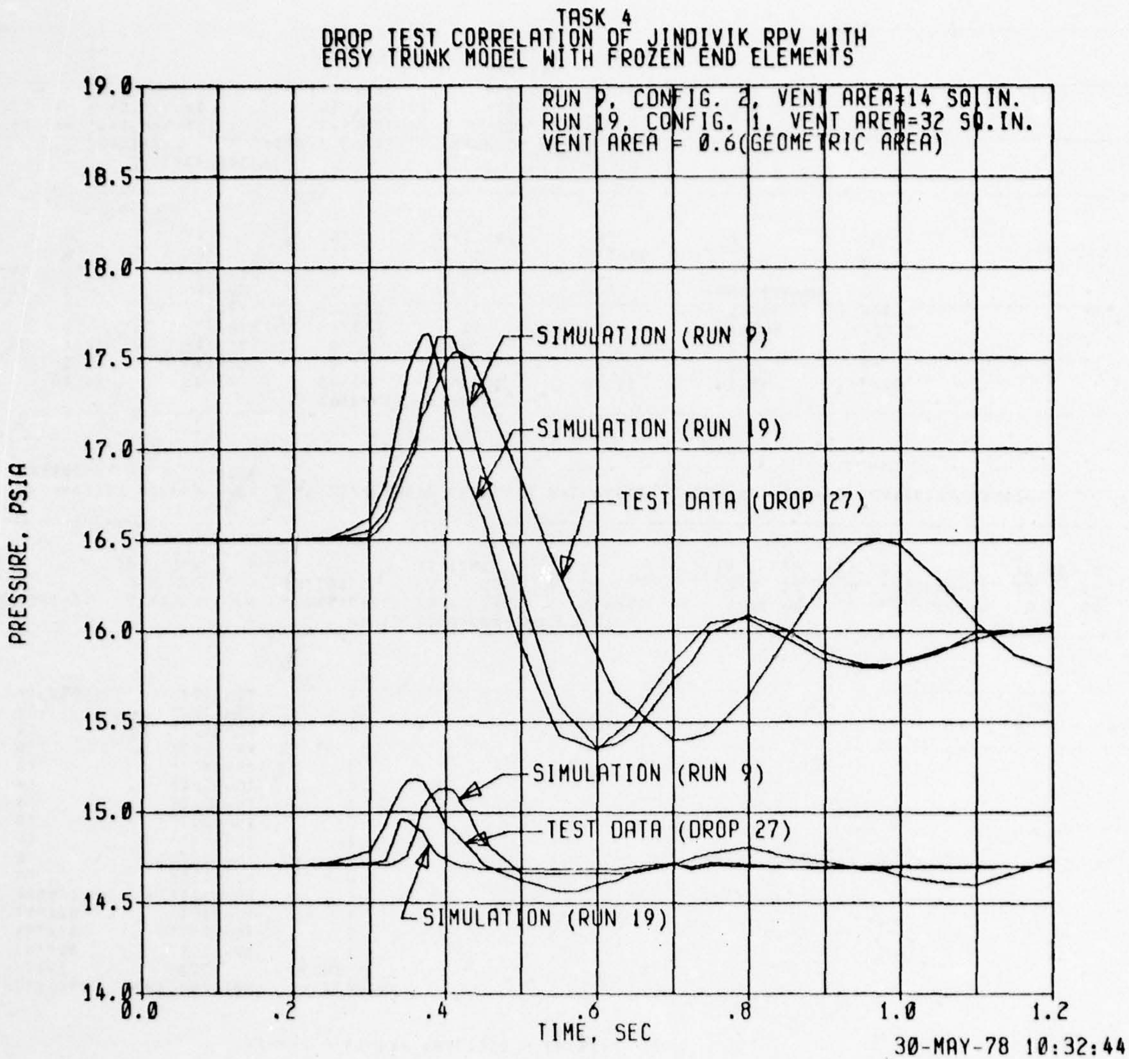


Figure 51 Trunk Model with Frozen End Elements

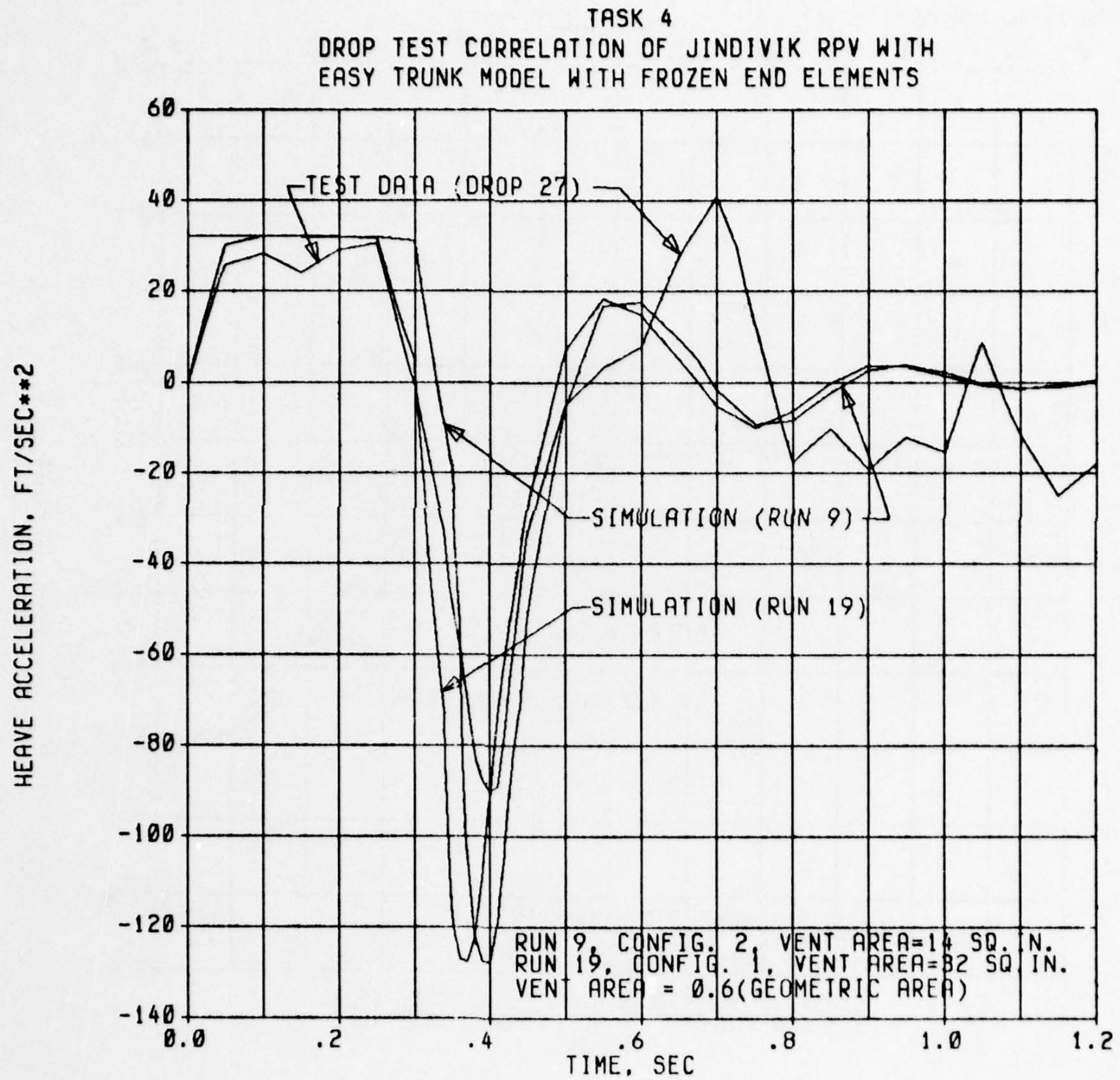


Figure 52 Trunk Model with Frozen End Elements

25-MAY-78 15:35:06

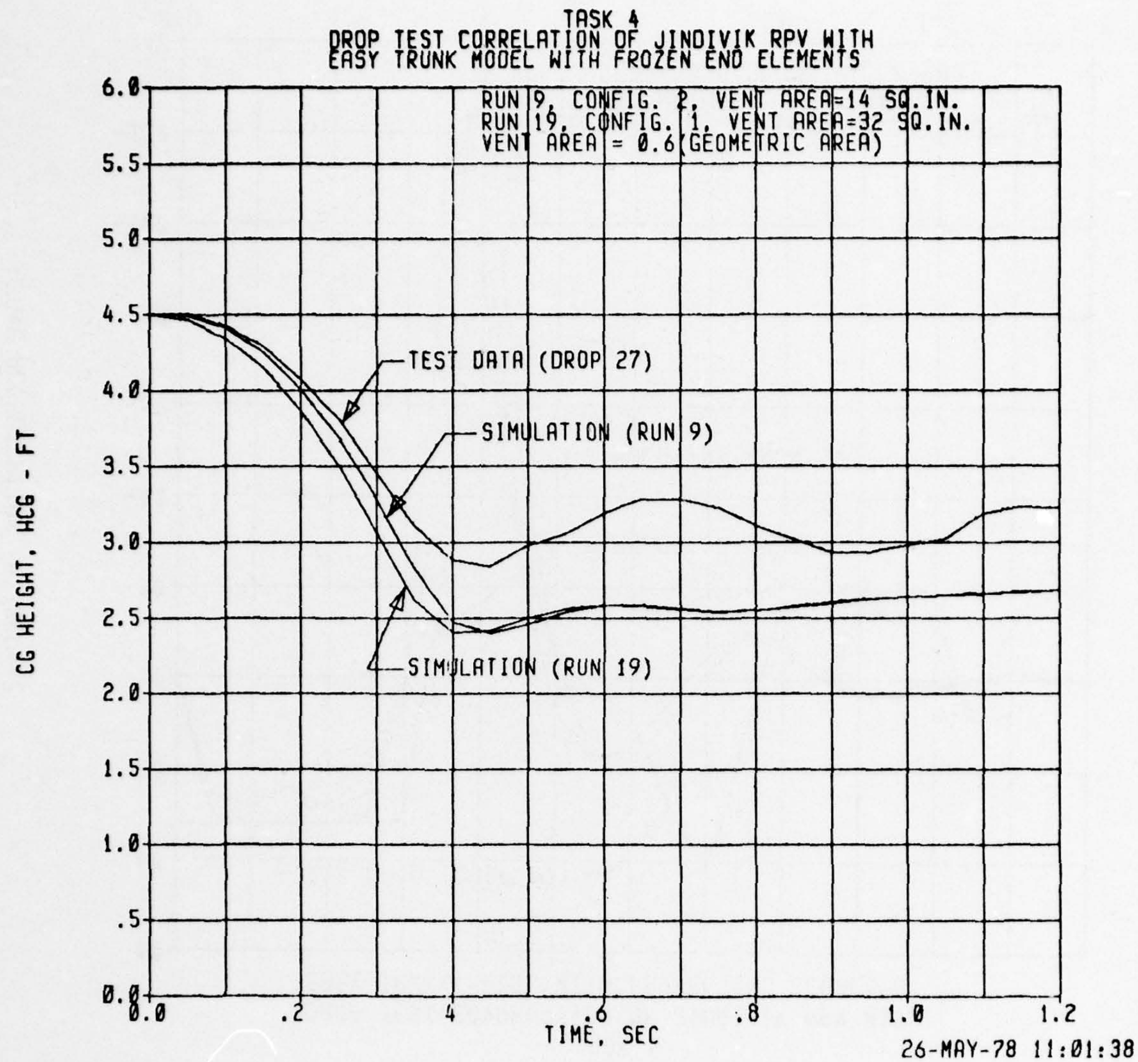


Figure 53 Trunk Model with Frozen End Elements

TABLE 3 SIMULATION RUNS MATRIX

CONFIGURATION (RUN NO.)	TRUNK MODEL FOR END ELEMENTS (I TYPE)		CUSHION VENT OPEN AREA (CAC)	TRUNK DAMPING COEFF (DMP)	SIMULATION RESULTS				o/o DIFFERENCE FROM DROP TEST NO. 27 DATA			
					PT_{max}	PC_{max}	$-Gcg_{max}$	PLUNGE FREQ.	BASED ON GAGE PRESSURES		-Gcg	PLUNGE FREQ.
	FROZEN	CONSTRAIN MEMBRANE	IN^2	LB-SEC/ IN/IN^2	Psia (Psig)	Psia (Psig)	Ft/Sec^2	Rad/Sec	PT	PC		
1 (19)	•		32 (FULL OPEN)	.025	17.62 (2.92)	14.96 (0.26)	127	14.6	2.5	-41.0	40.0	39.5
2 (9)	•		14	.025	17.62 (2.92)	15.13 (0.43)	127	14.6	2.5	-2.3	40.0	39.5
3 (18)		•	32	.025	17.30 (2.60)	14.83 (0.13)	72	10.83	-8.8	-70.0	-21.0	3.4
4 (10)		•	14	.025	17.27 (2.57)	15.17 (0.47)	72	10.64	-9.8	6.8	-21.0	1.6
5 (17)	•	•	18	.020	17.30 (2.60)	15.16 (0.46)	76	10.83	-8.8	4.5	-16.0	3.4

DATA COMMON TO ALL RUNS:

- Flow rate into trunk (WTR) = 48 lb/min
- All side elements simulated as Membrane Models
- Relief Valve Data: see Figure 1
  - P_{crack} = 1.94 psig (area = 0)
  - $P_{full-open}$ = 2.77 psig (area = 32.4 in^2)
- Coeff. of discharge for cushion vent = 0.6 (53.3 x 0.6 = 32) = Effective area

DROP TEST NO. 27 DATA:

- PT_{max} = 17.55 psia (2.85 psig)
- PC_{max} = 15.14 psia (0.44 psig)
- $-Gcg_{max}$ = 90.8 Ft/Sec^2
- Trunk Plunge Freq. = 10.47 rad/sec

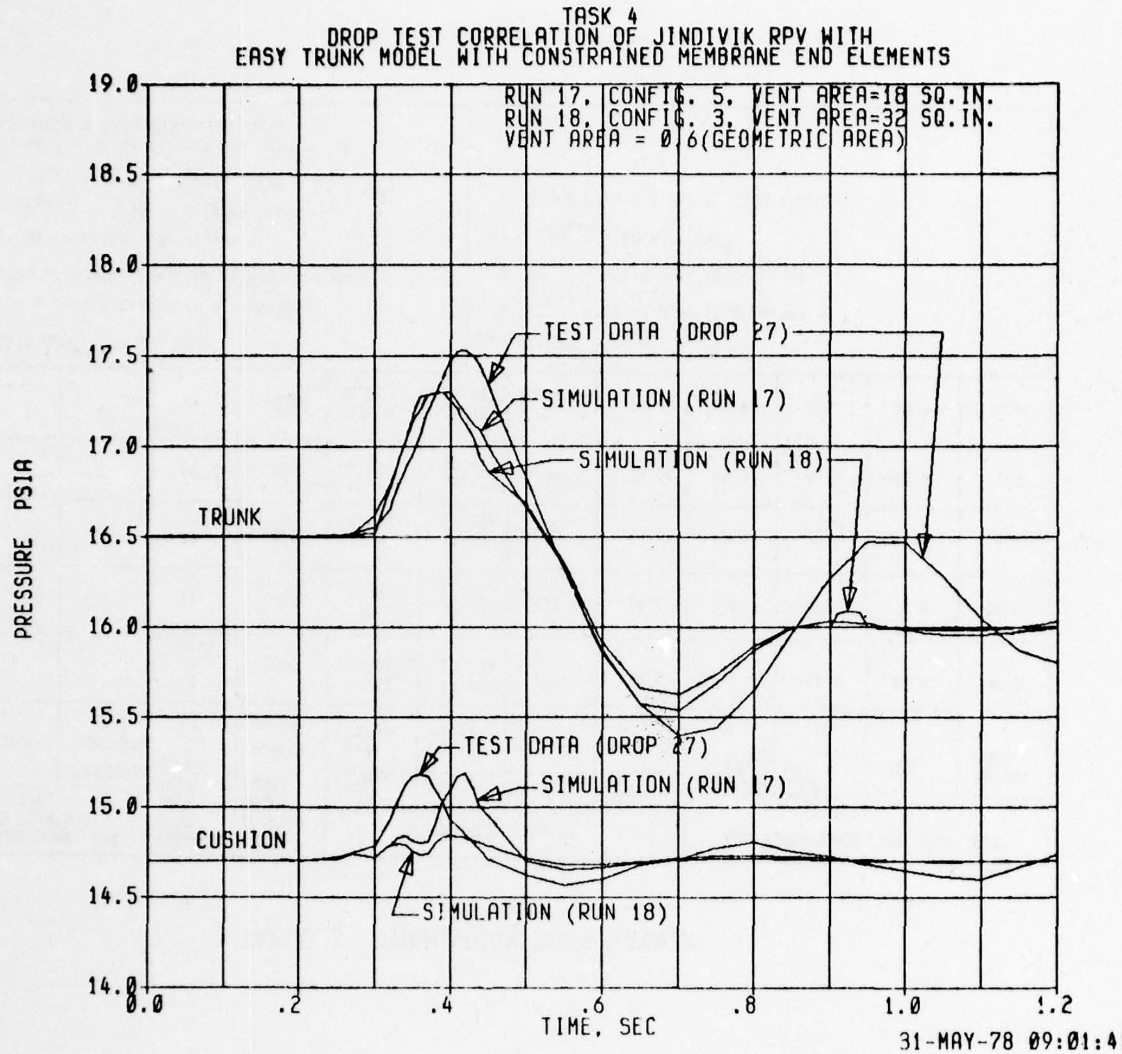


Figure 54 Trunk Model with Constrained Membrane End Elements

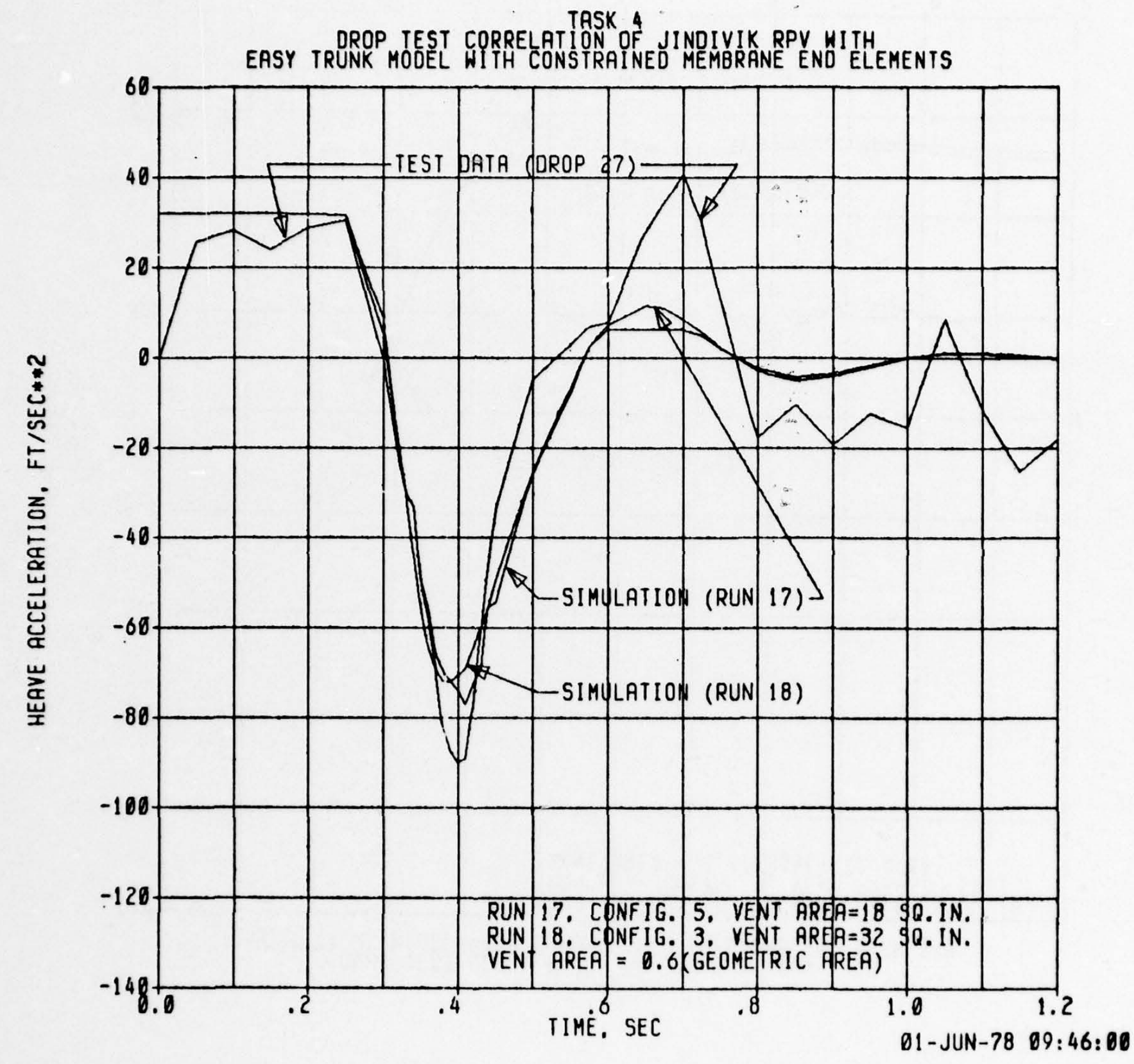


Figure 55 Trunk Model with Constrained Membrane End Elements

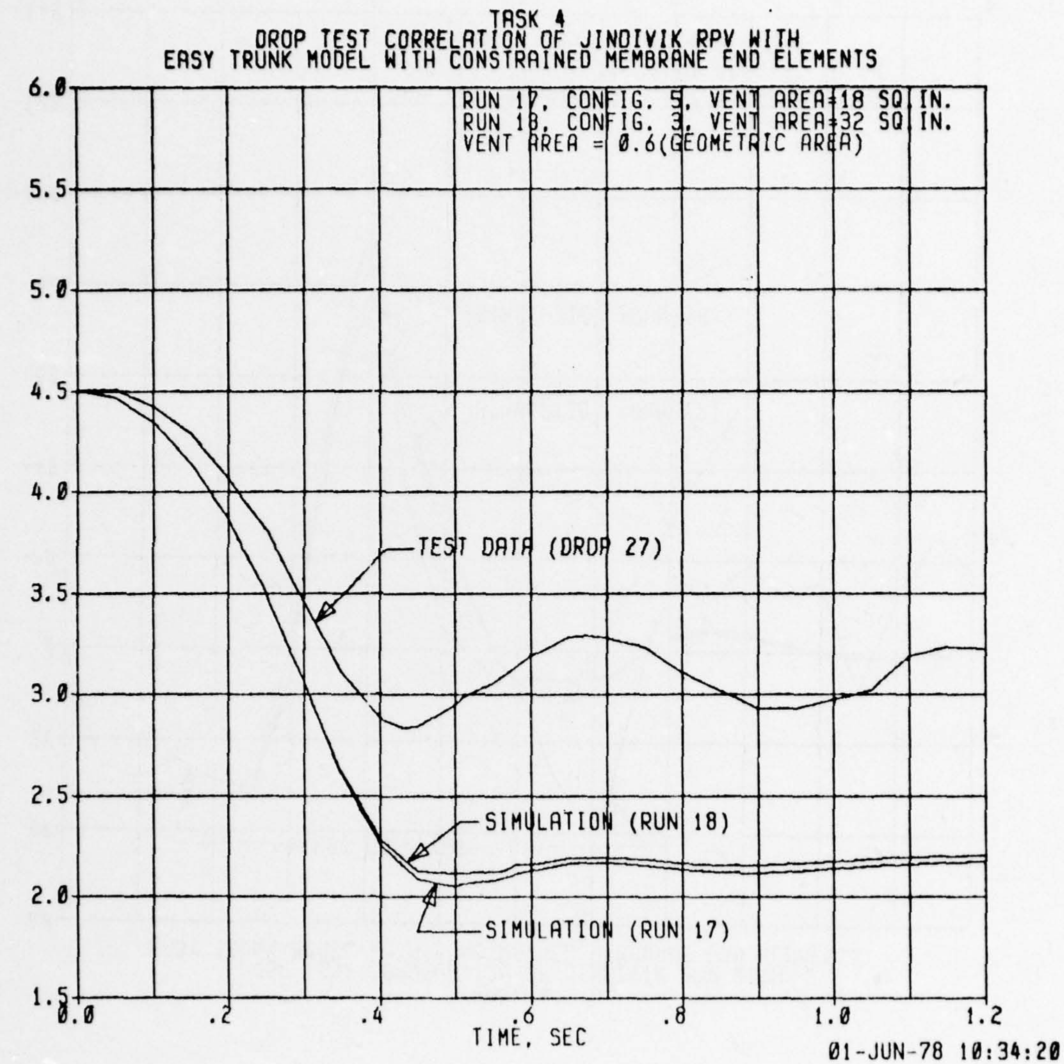


Figure 56 Trunk Model with Constrained Membrane End Elements

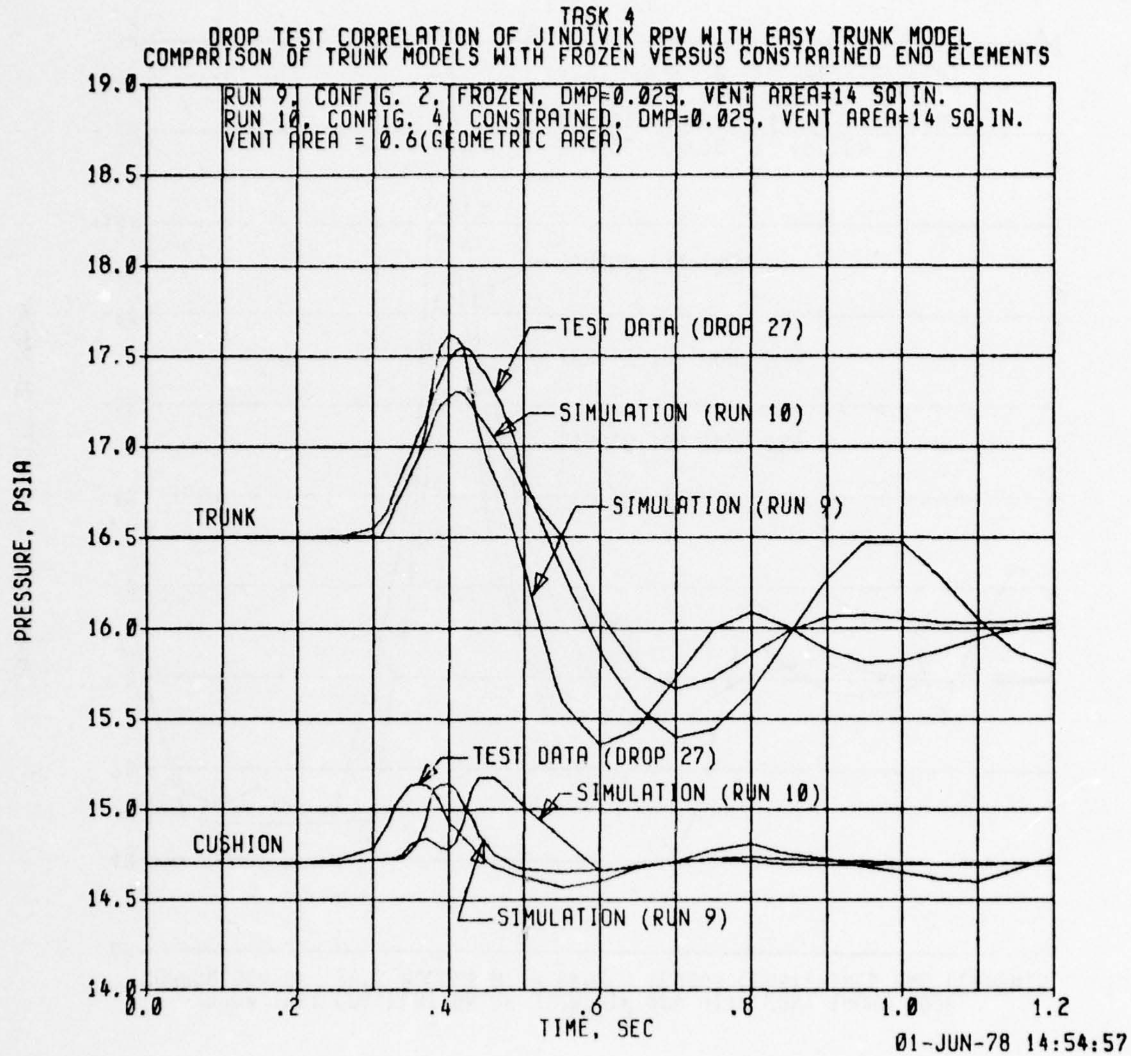


Figure 57 Comparison of Trunk Models, Frozen vs Constrained End Elements

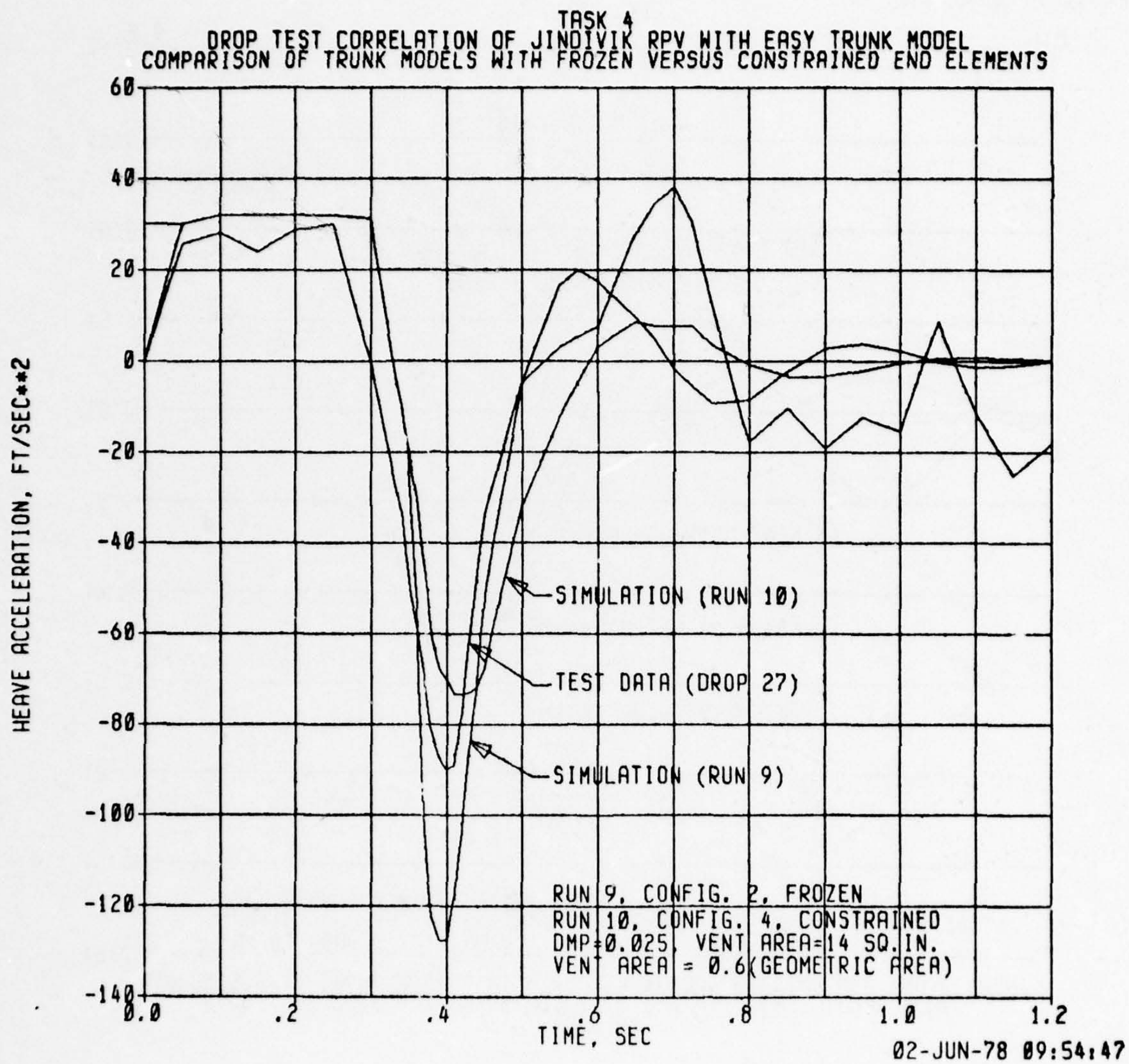


Figure 58 Comparison of Trunk Models, Frozen vs Constrained End Elements

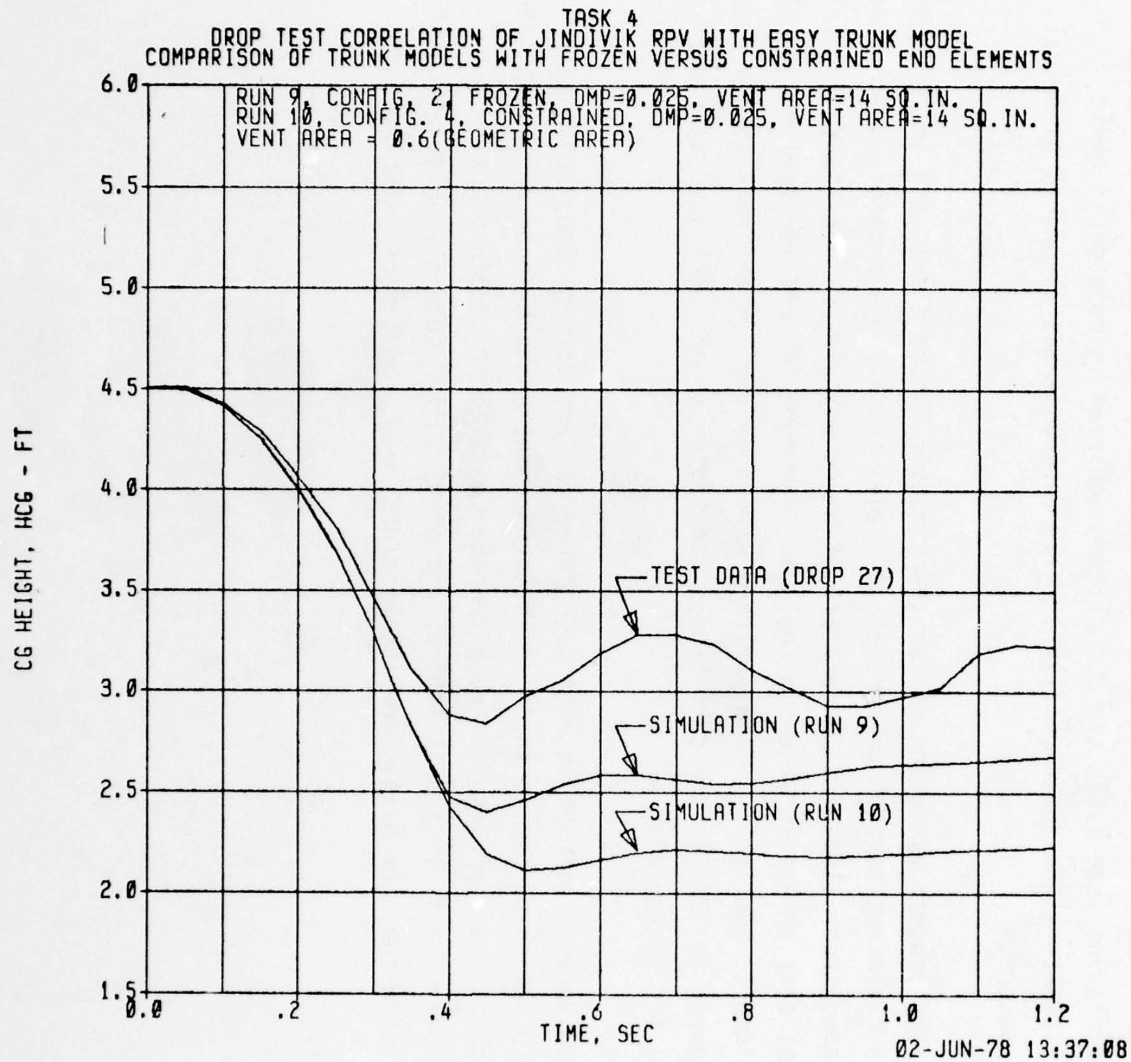


Figure 59 Comparison of Trunk Models, Frozen vs Constrained End Elements

3.4 Discussion

In the light of difficulties with the type of drop test data available for correlation (see 3.2), the correlation achieved in considered quite satisfactory and the objective of ±10% tolerance was abandoned by mutual agreement.

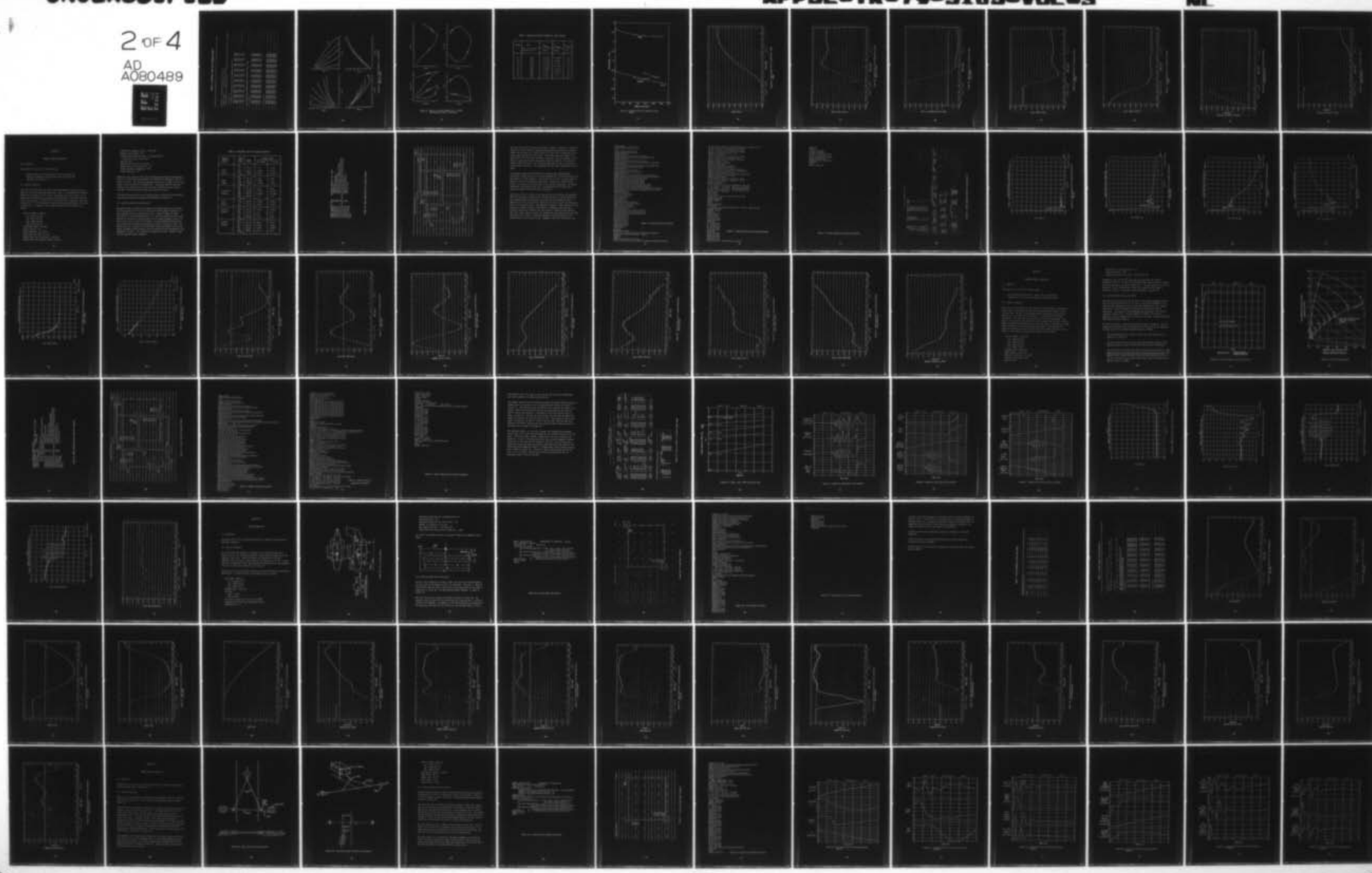
During program execution of the trunk module TK the first step is to calculate trunk element section properties or Digges trunk shape parameters for free and loaded trunk elements. An example of these calculated (and stored for later use) data arrays is shown in Table 4. For more information/explanation see Volume I, Section IV. These arrays were then plotted and results are shown in Figures 60 and 61; the data arrays for frozen trunk model used in Figure 61 are not included here.

From various simulation runs made for drop test correlation, pertinent information on peak cushion and trunk pressures has been extracted for several cushion vent opening and trunk damping coefficient combinations. This information is listed in Table 5 and plotted in Figure 62. Figures 63 through 69 show the remaining time-history plots requested for the simulation.

AD-A080 489 BOEING AEROSPACE CO SEATTLE WA BOEING MILITARY AIRPL--ETC F/G 9/2
EASY ACLS DYNAMIC ANALYSIS. VOLUME III. DESCRIPTION OF SIMULATI--ETC(U)
SEP 79 M K WAHI, P R PERKINS, G S DULEBA F33615-77-C-3054

UNCLASSIFIED AFFDL-TR-79-3105-VOL-3 NL

2 OF 4
AD
A080489



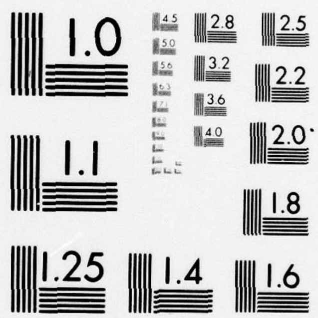

1.0
1.1
1.25
1.4
1.6
1.8
2.0
2.2
2.5
2.8
3.2
3.6
4.0
4.5
5.0
5.6
6.3

TABLE 4 CALCULATED TRUNK SHAPE PARAMETER ARRAYS

••••• TRUNK ELEMENT SECTION PROPERTIES••

••• DATA SET 2 ••• MEMBRANE TRUNK ELEMENT ••••••••••••••••

NPTS=10 DPR=.100 A= 9.33 B= 5.60 LO= 55.00

ELEMENT PROPERTIES FREE OF GROUND EFFECTS AND PR=0 ZO= 15.55 YO= 9.33 L1= 33.22 ACV= 138.4 AS= 313.9

••• ZO ARRAY •••

1ST COLUMN IS OUT-OF-GROUND-EFFECTS ZO FOR INCREASING VALUES OF PR

15.55	13.99	12.44	10.88	9.33	7.77	6.22	4.66	3.11	1.55
14.76	13.28	11.81	10.33	8.86	7.38	5.90	4.43	2.95	1.48
13.76	12.39	11.01	9.63	8.26	6.88	5.50	4.13	2.75	1.38
12.55	11.30	10.04	8.79	7.53	6.28	5.02	3.77	2.51	1.26
11.14	10.03	8.92	7.80	6.69	5.57	4.46	3.34	2.23	1.11
9.54	8.59	7.63	6.68	5.73	4.77	3.82	2.86	1.91	.95
7.77	7.00	6.22	5.44	4.66	3.89	3.11	2.33	1.55	.78
5.87	5.28	4.70	4.11	3.52	2.93	2.35	1.76	1.17	.59
3.89	3.50	3.11	2.72	2.33	1.94	1.55	1.17	.78	.39
1.90	1.71	1.52	1.33	1.14	.95	.76	.57	.38	.19

THE FOLLOWING ARRAYS CORRESPOND ELEMENT-TO-ELEMENT WITH THE ZO ARRAY

••• YO ARRAY •••

9.33	9.00	8.86	8.78	8.67	8.43	8.01	7.34	6.31	4.68
10.86	10.47	10.19	9.92	9.59	9.16	8.57	7.75	6.59	4.84
12.31	11.84	11.41	10.57	10.47	9.86	9.11	8.15	6.86	5.00
13.66	13.09	12.53	11.93	11.27	10.51	9.61	8.52	7.11	5.14
14.85	14.19	13.50	12.78	11.98	11.09	10.06	8.84	7.33	5.26
15.87	15.11	14.32	13.48	12.57	11.57	10.44	9.12	7.52	5.36
16.66	15.83	14.95	14.03	13.03	11.94	10.72	9.33	7.65	5.44
17.17	16.29	15.36	14.37	13.32	12.17	10.90	9.45	7.73	5.47
17.37	16.46	15.51	14.50	13.41	12.24	10.94	9.47	7.73	5.46
17.22	16.32	15.36	14.37	13.29	12.12	10.84	9.38	7.65	5.41

••• L1 ARRAY •••

33.22	32.99	33.11	33.47	34.00	34.62	35.30	36.02	36.77	37.54
33.70	33.60	33.73	34.04	34.48	35.00	35.58	36.22	36.89	37.59
34.16	34.15	34.30	34.57	34.93	35.37	35.87	36.43	37.02	37.65
34.62	34.67	34.82	35.06	35.37	35.75	36.17	36.65	37.17	37.72
35.08	35.17	35.32	35.53	35.80	36.12	36.46	36.86	37.32	37.80
35.55	35.65	35.80	35.99	36.22	36.48	36.76	37.12	37.48	37.88
36.02	36.13	36.27	36.44	36.63	36.85	37.09	37.36	37.65	37.96
36.52	36.63	36.75	36.89	37.04	37.21	37.40	37.60	37.81	38.04
37.06	37.15	37.24	37.34	37.46	37.57	37.70	37.84	37.98	38.13
37.64	37.69	37.75	37.81	37.87	37.93	38.00	38.07	38.14	38.21

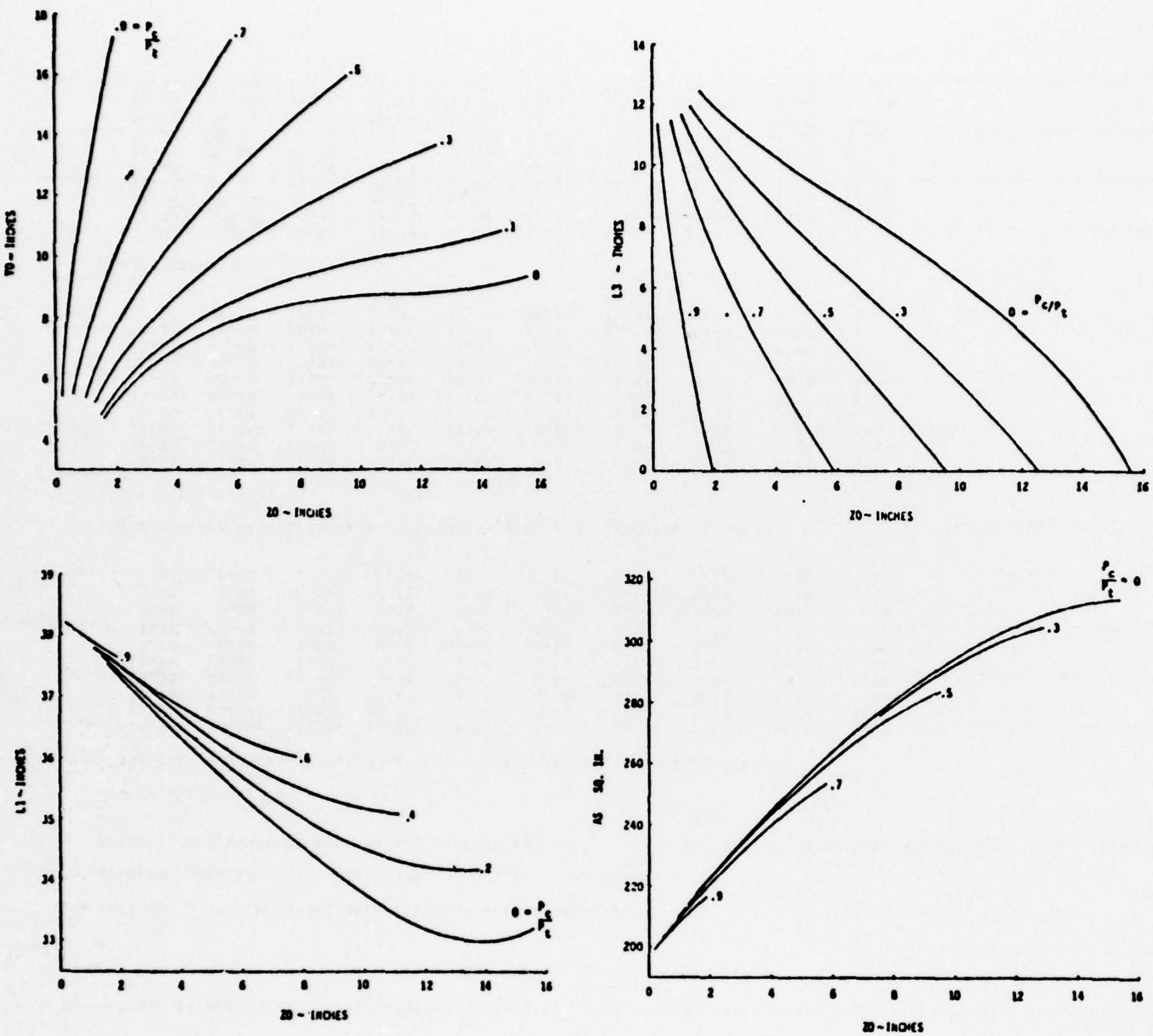


Figure 60 ACRS No. 2 Side Element No. 5, Membrane Model (Loaded)

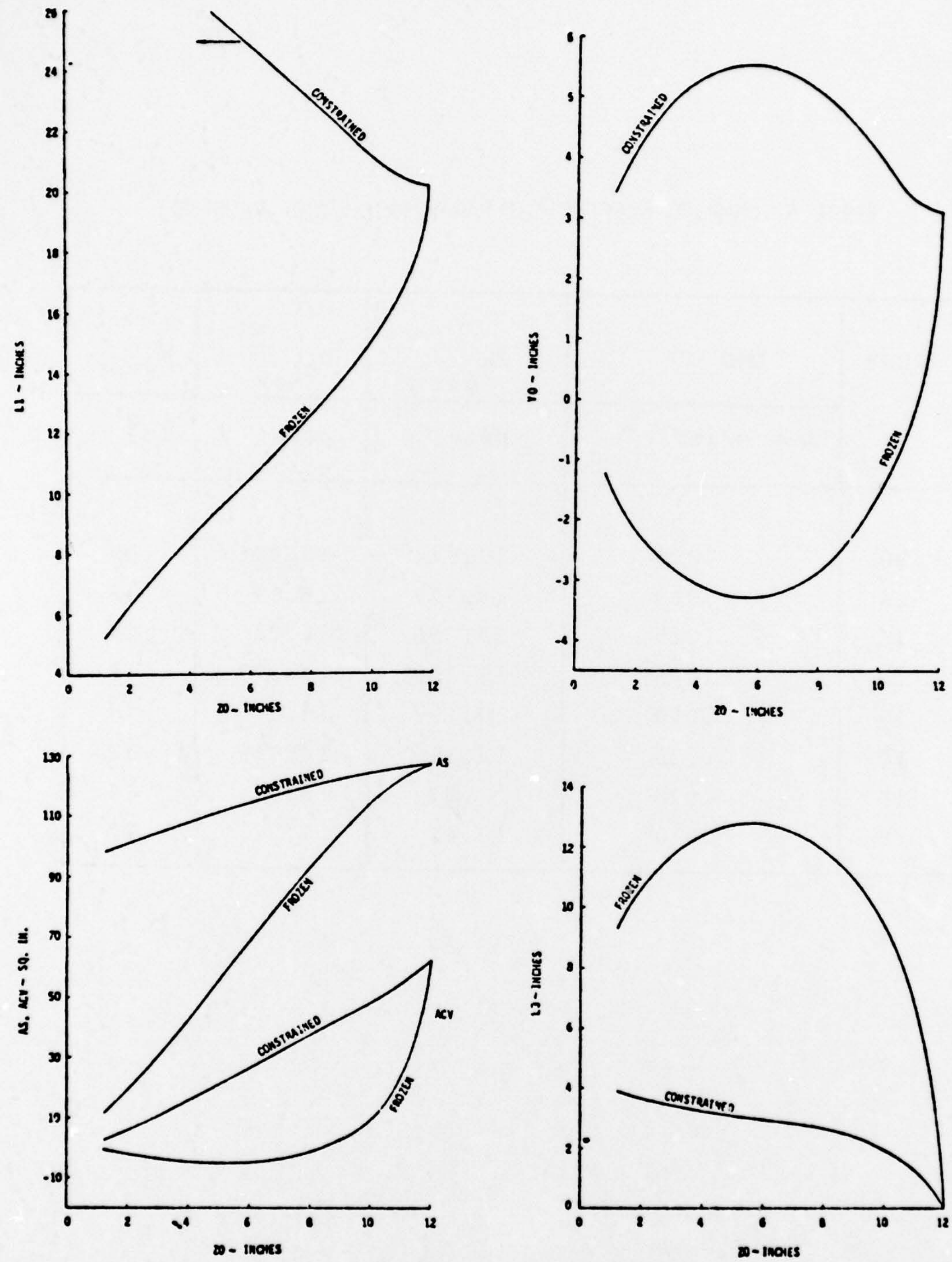


Figure 61 ACRS No. 2 Front Element No. 1, Frozen and Contrained Models Comparison

TABLE 5 DAMPING COEFFECIENT PARAMETRIC STUDY RESULTS

RUN#	DMP	$P_{T_{max}}$	$P_{c_{max}}$	A_{vent}
	Lb-sec/in/in^2	psia	psia	in^2
20	.000	17.674	15.404	29
14	.010	17.378	15.09	32
16	.015	17.356	15.03	29
12	.015	17.338	15.377	18
17	.020	17.298	15.16	18
18	.025	17.30	14.835	32
10	.025	17.27	15.17	14
15	.010	17.42	15.22	27

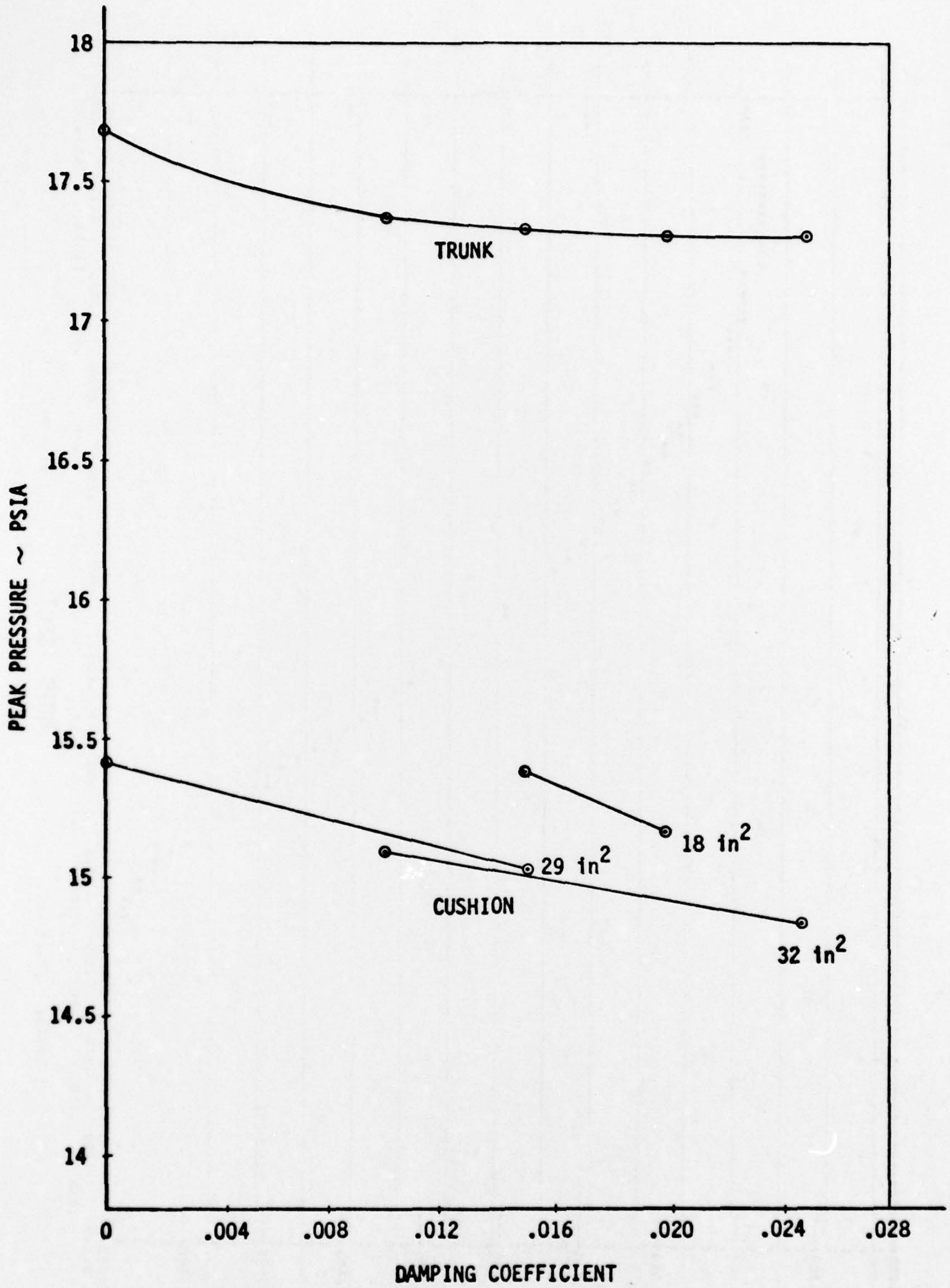


Figure 62 Damping Coefficient Parametric Study Results

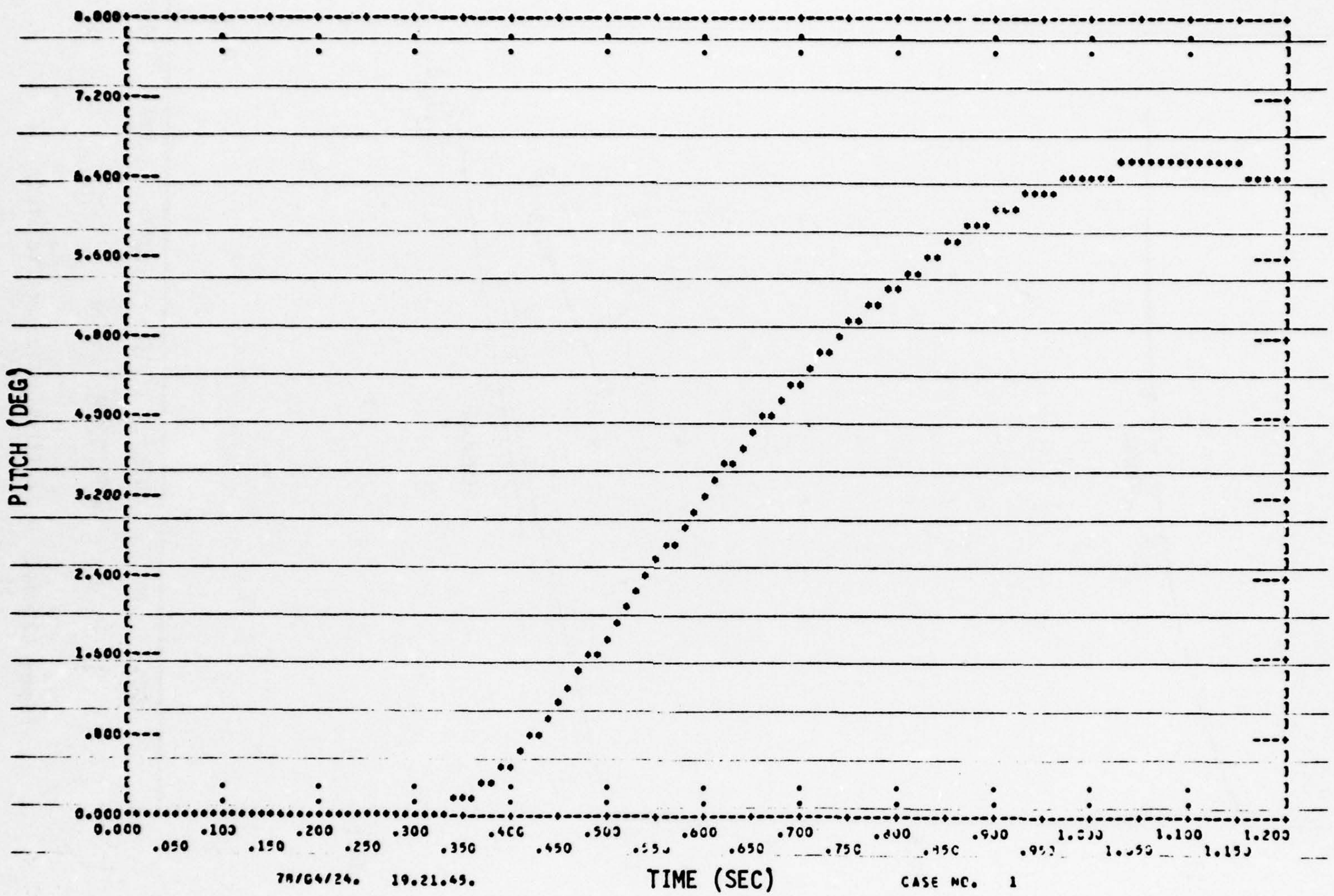


Figure 63 Time History Plot of Drop Test, Pitch Angle

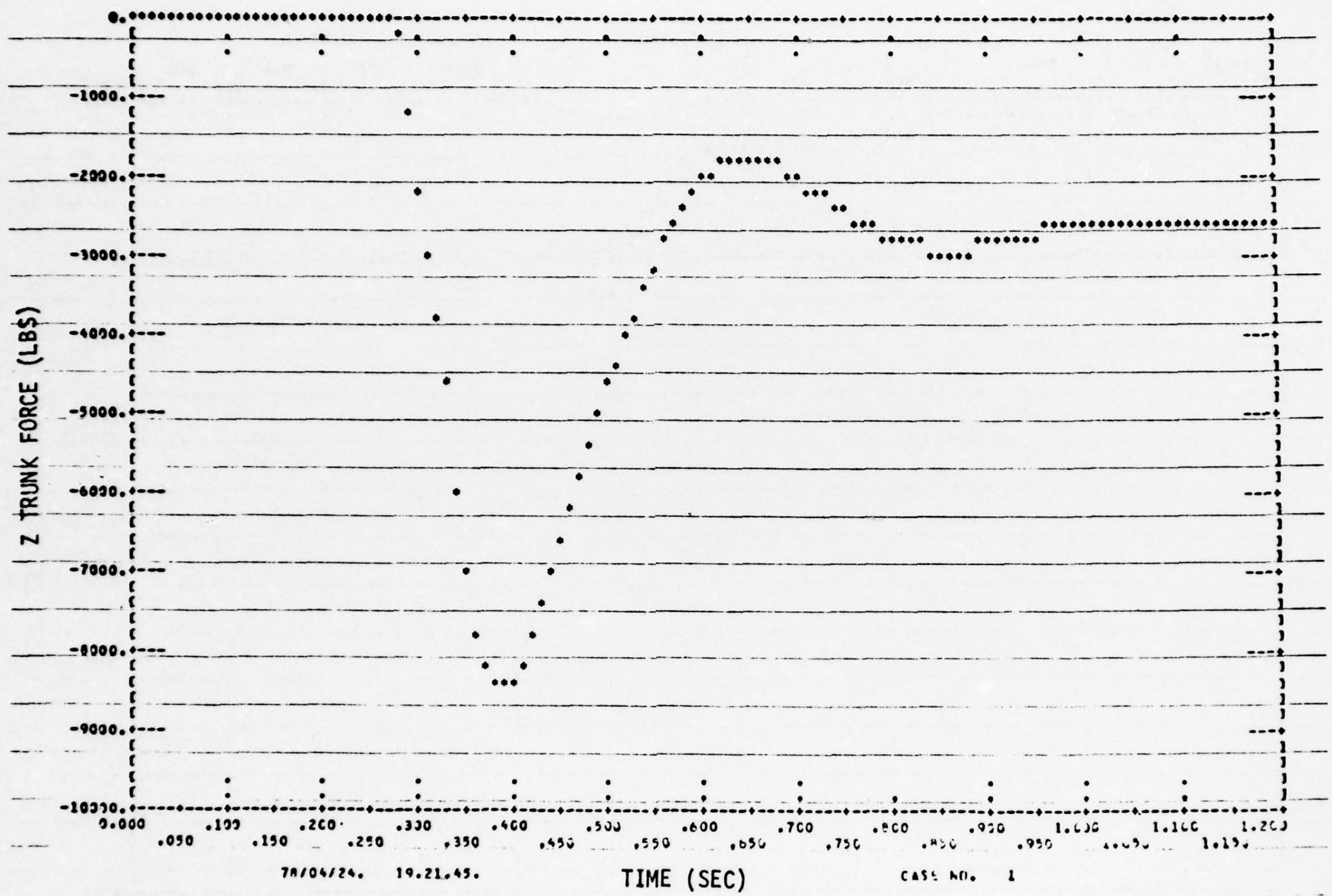


Figure 64 Time History Plot of Drop Test, Trunk Vertical Force

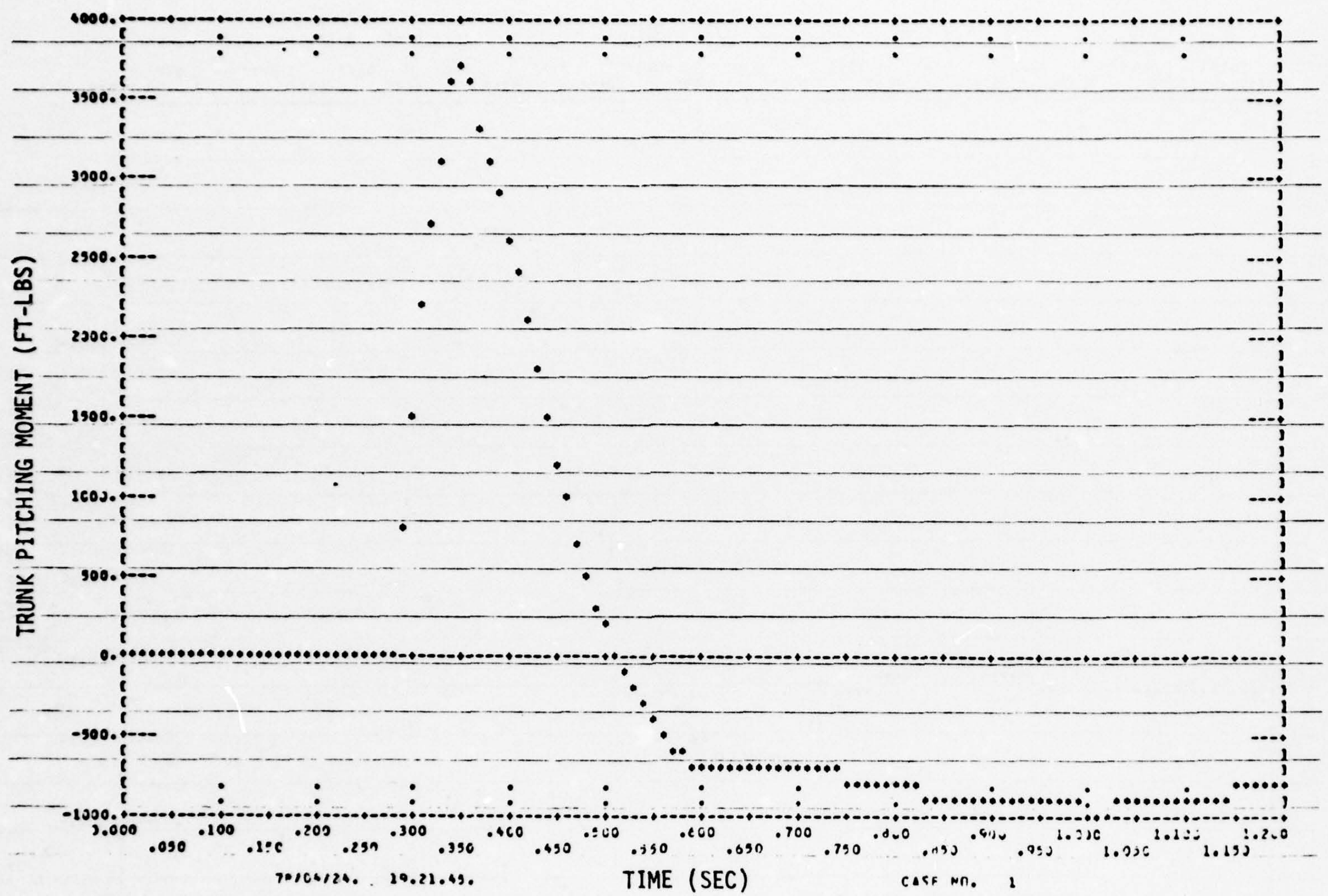


Figure 65 Time History Plot of Drop Test, Trunk Pitch Moment

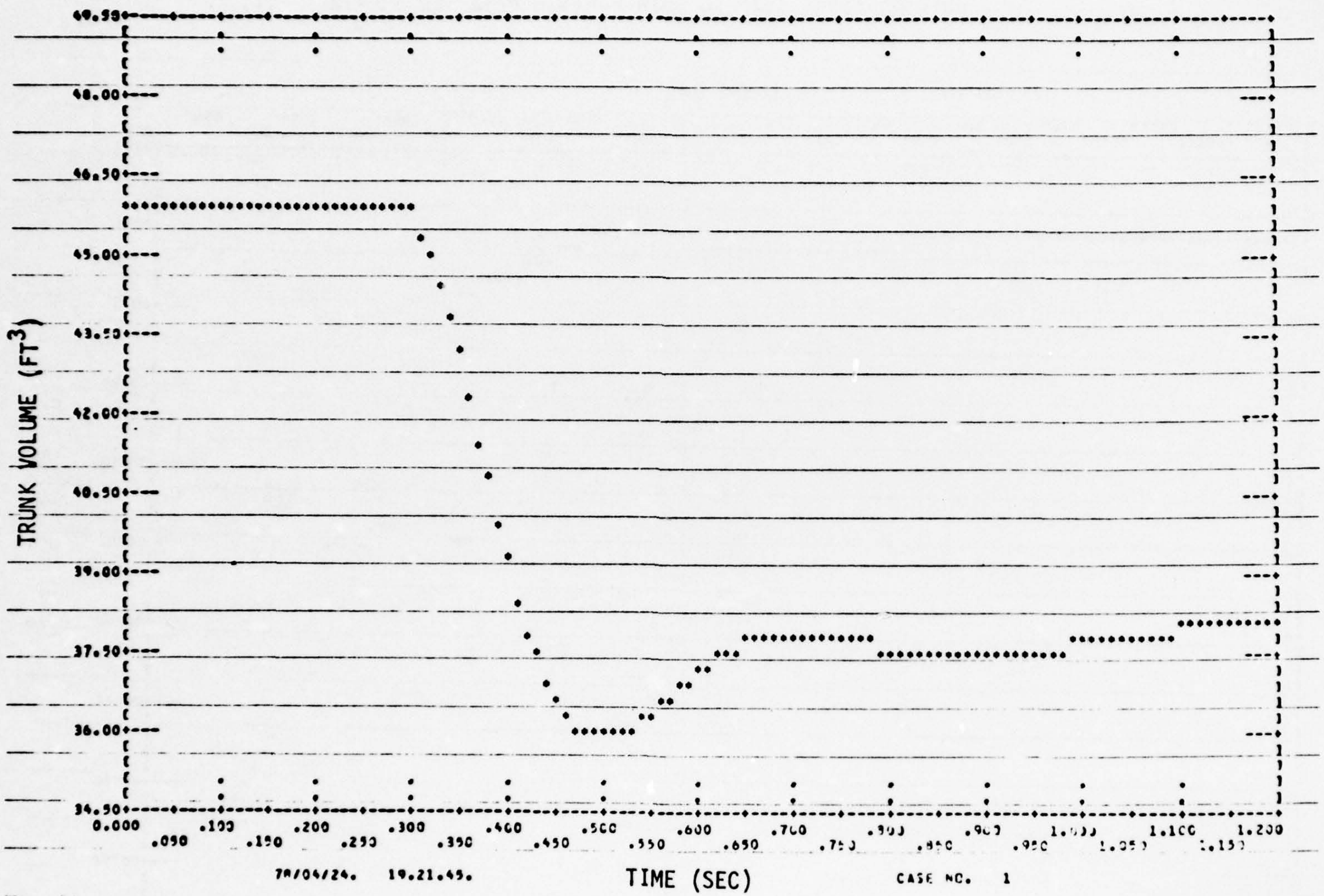


Figure 66 Time History Plot of Drop Test, Trunk Volume

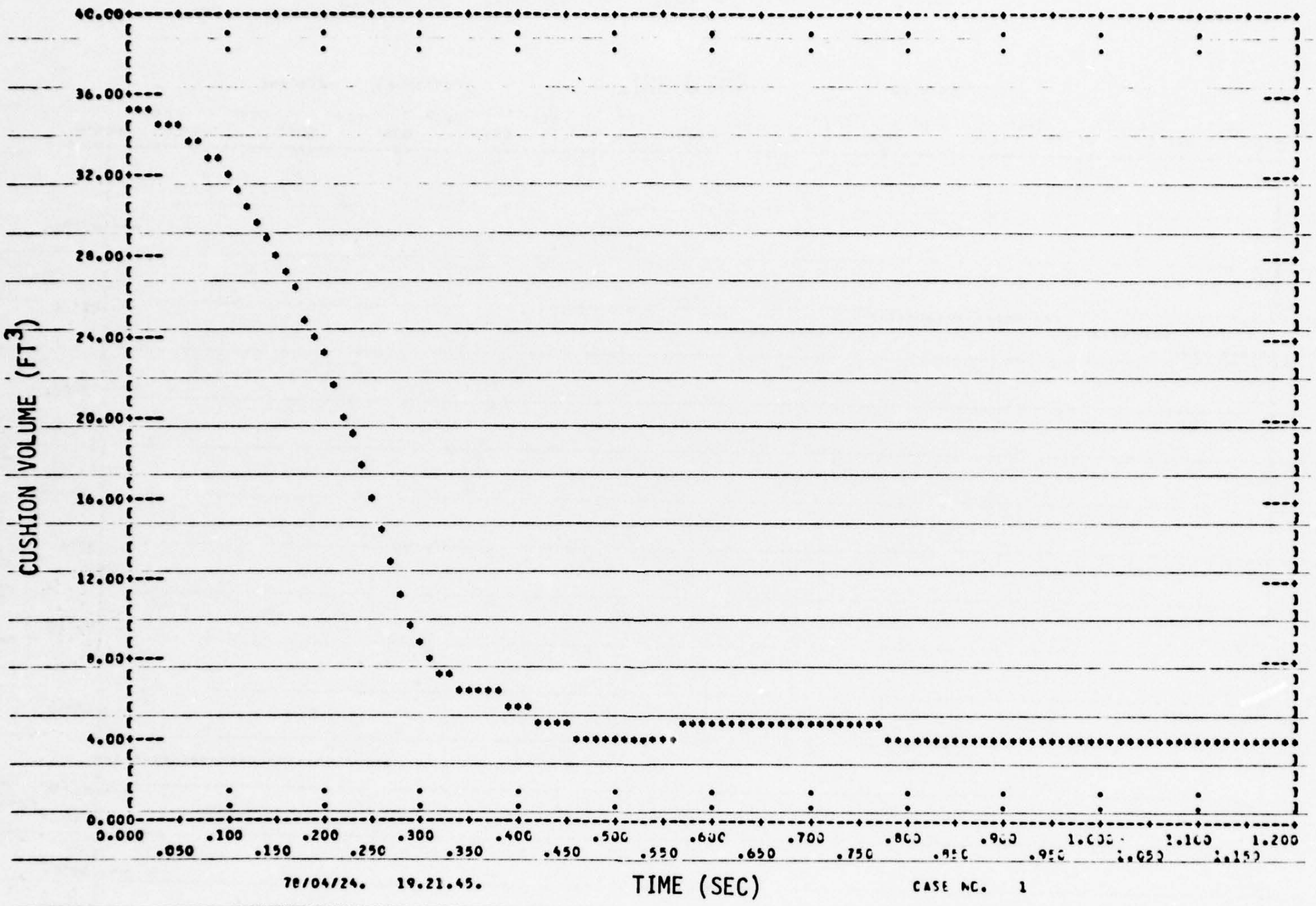


Figure 67 Time History Plot of Drop Test, Cushion Volume

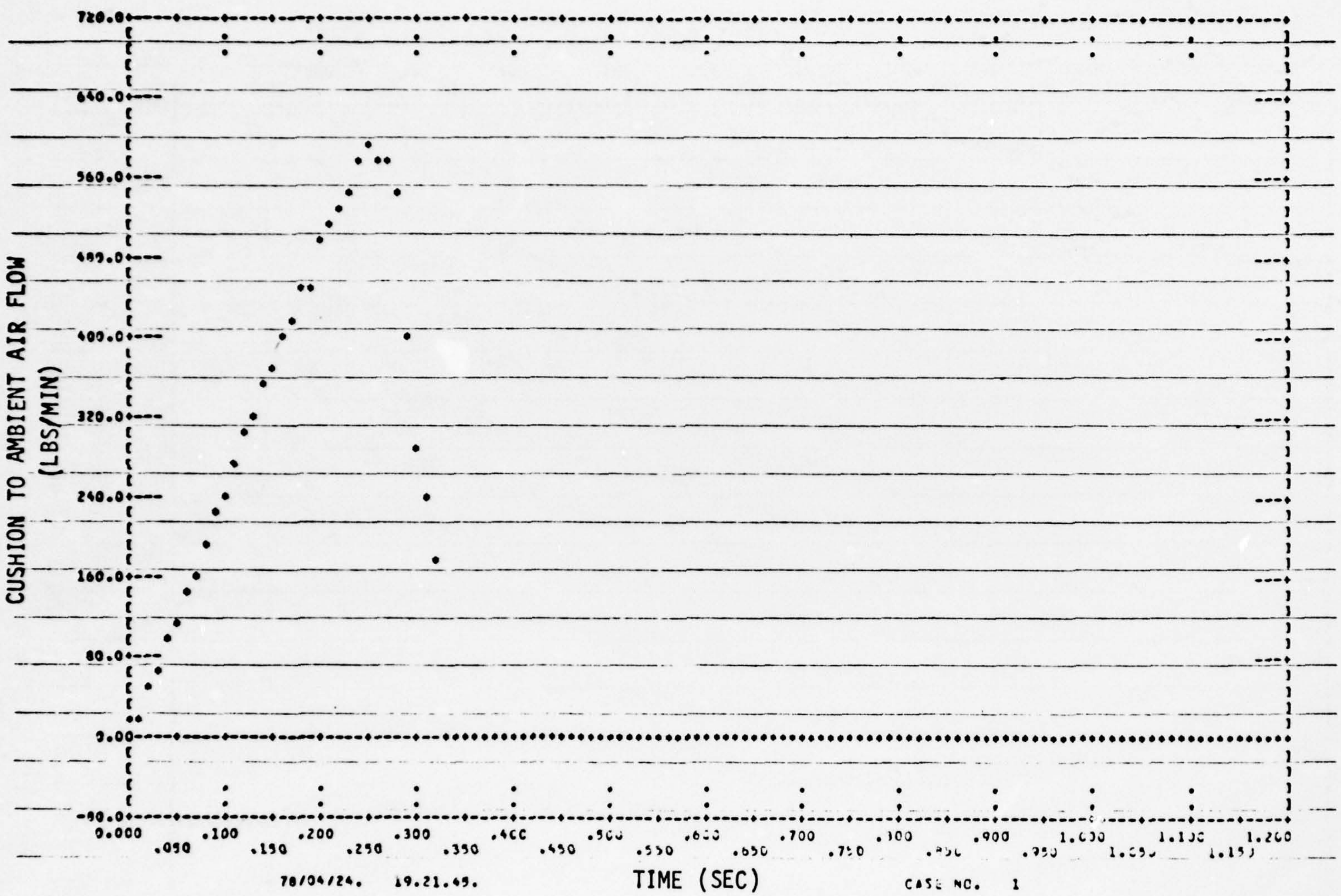


Figure 68 Time History Plot of Drop Test, Cushion to Ambient Airflow

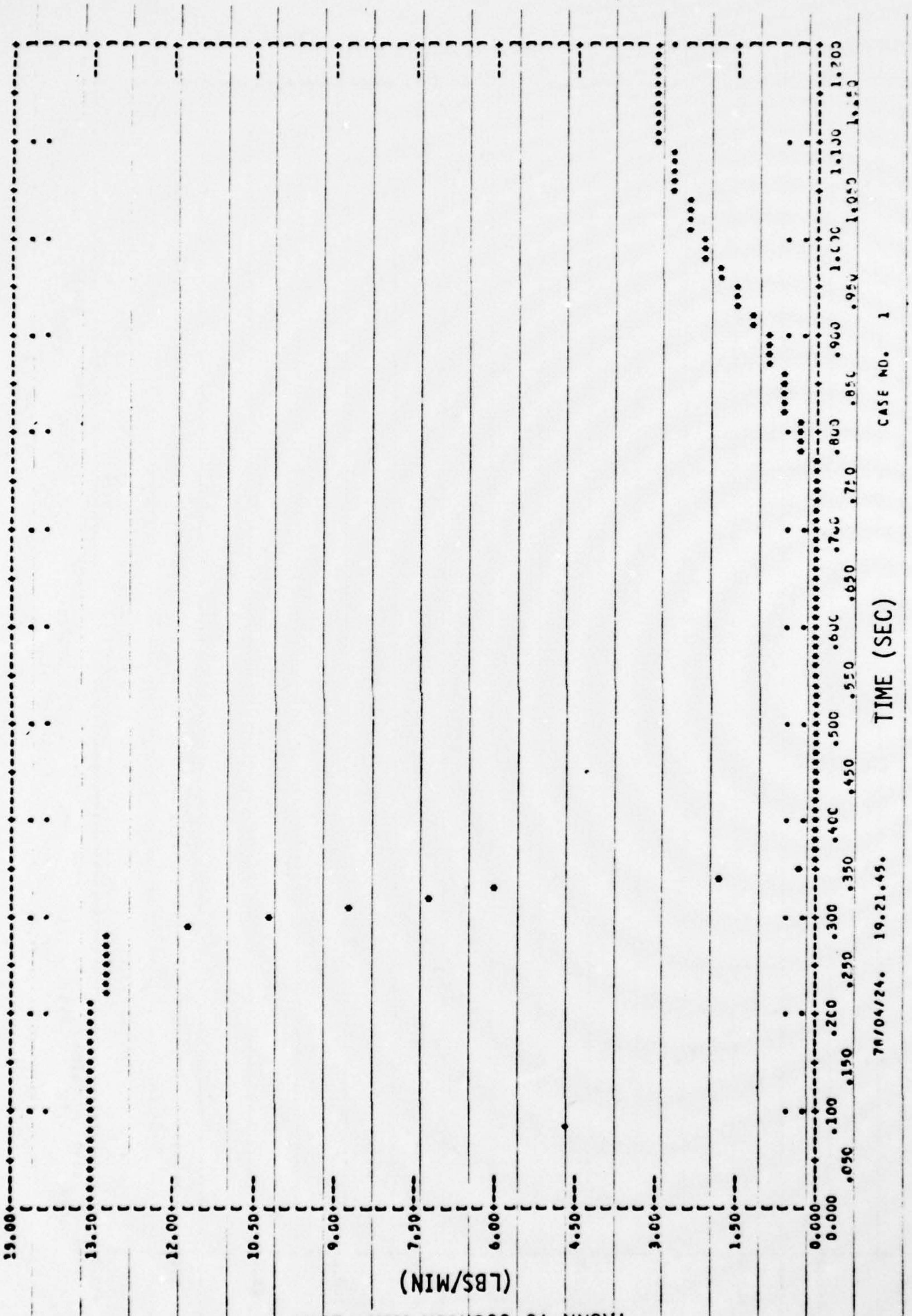


Figure 69 Time History Plot of Drop Test, Trunk to Cushion Airflow

SECTION IV

JINDIVIK LANDING SIMULATION

4.1 Objectives

The objective of this Task 4 simulation was:

- Using components developed in Tasks 1 and 3 simulate the approach, touchdown, and slideout of the Jindivik on the recovery trunk ACRS No. 2.

4.2 Technical Approach

The 6 DOF rigid body model was used. The trunk geometric data used was the same as that for the drop test simulation, Task 4. The approach and touchdown phases were simulated with 19^{o} flaps. The steady state (trim) condition for approach was determined by using the optimal controller in the same fashion as it was used for Task 2 inflight trim. 0.3 seconds after touchdown, flaps were retracted to 1^{o} for the remaining slideout and engine was allowed to spin down to idle thrust. A summary of the assumptions made for the analysis is as follows:

Aircraft weight = 2900 lbs.
I_{xx} = 1190 slug ft^2
I_{yy} = 1810 slug ft^2
I_{zz} = 2840 slug ft^2
I_{xz} = -200 slug ft^2
Touch down speed = 130 knots
Pitch attitude = $.4^{o}$
Rate of descent = 7.5 ft/sec
Flap retraction time = 0.3 sec
Engine idle thrust (slide out) = 300 lbs.
Engine thrust for approach (Trim) = 740 lbs.

Elevator for approach (Trim) = -6.6^{o} (up)

Elevator for slideout = 0^{o}

Damping coefficient for Trunk = .02 $lb\text{-}sec/in/in^2$

(same as for drop test simul.)

ACLS Fan OFF

Coefficient of friction for front = .2

Coefficient of friction for aft = .8

Flaps (approach & touchdown) = 19^{o}

Flaps (slide out) = 1^{o}

Aerodynamic data used for these flap settings were extracted from Reference 5, Section 4, "Wind Tunnel Data Analysis" and Reference 6, Appendix A, and are shown in Table 6. The lateral aerodynamic derivatives in Table 6 are for wind tunnel test configuration 4 (Extended fin). These were used for stability reasons as the configuration 3 data indicates instability of the vehicle in that mode and would have complicated the simulation results.

The engine characteristics and the pressure drop K factors for the airflow to trunk supply system were extracted from Reference 5, Section 2.

4.3 Simulation Description and Results

The modules used to simulate the landing and slideout dynamics are the basic airplane components SG, VA, OL, DL and ES; the trunk-cushion component TK, a duct component DU and a summing junction S2. Figure 70 shows the model description for the landing simulation. For the trim-calculation part of the simulation, the optimal controller OC will be an added component and will be used in the loop as in Section 2 example. The duct component DU represents the pressure drop from the engine bleed port outlet to the trunk inlet. The summing junction is needed to add the engine forces and moments to the trunk forces and moments before they are input as external forces and moments to the components OL and DL. This completes the model definition. Figure 71 shows the EASY generated model schematic.

TABLE 6 AERODYNAMIC DATA FOR LANDING SIMULATION

TYPE OF COEFFI-CIENT	TEXT BOOK NAME	EASY NAME	VALUE USED	
			19° FLAPS	1° FLAPS
LIFT FORCE	$-C_{L_o}$	ZO OL	-.52	-.145
	$-C_{L_\alpha}$	ZA OL	-4.584	-4.04
	$-C_{L_{\delta_e}}$	ZDEOL	-.2674	-.2674
DRAG FORCE	$-C_{D_o}$	XO OL	-.165	-.09
	$-C_{D_\alpha}$	XA OL	-.7385	-.26
	$-C_{D_{\delta_e}}$	XDEOL	-.0764	-.0764
PITCHING MOMENT	C_{M_o}	MO OL	-.007	.056
	C_{M_α}	MALOL	-.1433	-.2195
	C_{M_q}	MQ OL	-6.1884	-6.1884
	$C_{M_{\delta_e}}$	MDEOL	-.72	-.72
SIDE FORCE	C_{y_β}	YB DL	-.8423	-.8423
	C_{y_p}	YP DL	-.0215	-.0215
ROLLING MOMENT	C_{l_β}	LB DL	-.1427	-.1427
	C_{l_p}	LP DL	-.37	-.37
	C_{l_r}	LR DL	.437	.437
	$C_{l_{\delta_a}}$	LDADL	-.187	-.187
YAWING MOMENT	C_{n_β}	NB DL	.0974	.0974
	C_{n_p}	NP DL	-.0457	-.0457
	C_{n_r}	NR DL	-.1545	-.1545
	$C_{n_{\delta_a}}$	NDADL	.0133	.0133

```
MODEL DESCRIPTION        JINDVIK LANDING TEST CASE
LOCATION=68     VA         INPUTS=SG
LOCATION=64     ES         INPUTS=VA(MAC=AMN)
LOCATION=52     DU         INPUTS=ES,TK(PT=P,2)
LOCATION=12     TK         INPUTS=SG,DU(W,2=WTR,T,2=TTR)
LOCATION=34     S2         INPUTS=ES(FX=FX,1,FZ=FZ,1,TY=TY,1)
TK(FXT=FX,2,FYT=FY,2,FZT=FZ,2,TXT=TX,2,TYT=TY,2,TZT=TZ,2)
LOCATION= 5     OL         INPUTS=VA,S2(3,1)
LOCATION=38     DL         INPUTS=OL,VA,S2(3,1)
LOCATION=10     SG         INPUTS=OL,DL
END OF MODEL
PRINT
```

Figure 70 Model Description for Landing Slideout

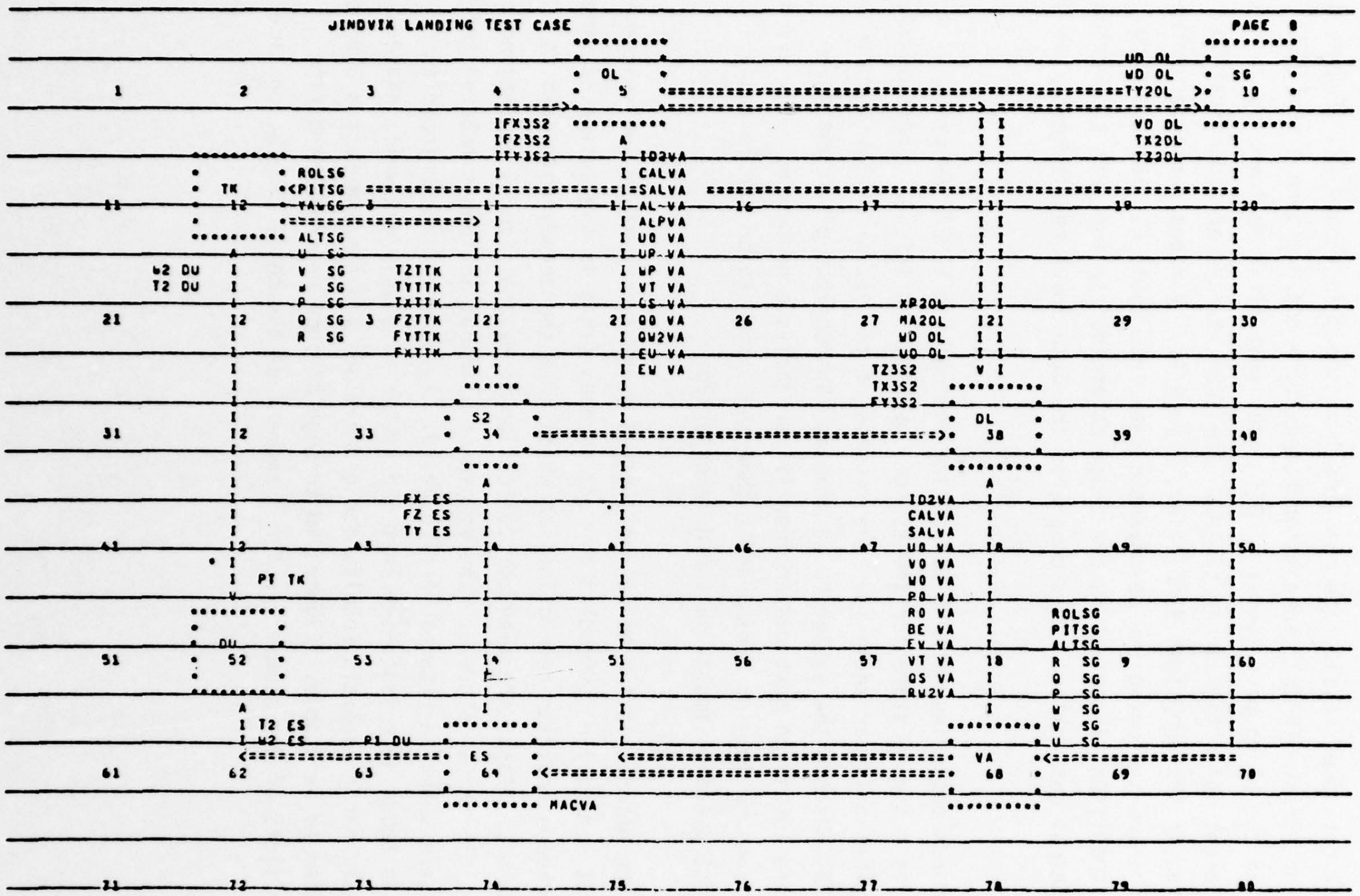


Figure 71 Jindivik Landing Simulation Model

The input data used to simulate this model is shown in Figure 72. Engine data tables for thrust (TSRES), compressor temperature and pressure rise (TBTES, TBPES) and the bleed port losses (TPOES) are input first followed by various trunk element definition array tables. The trunk arrays are identical to those used in the drop test simulation, Figure 42. Various parameter values are next input including the aerodynamic data from Table 6. The non-zero INITIAL CONDITIONS input values are those determined in the trim simulation using optimal controller. This completes the data input requirements for the model.

Thc commands beginning with NO STATES and ending with LINEAR ANALYSIS calculates a steady state for the system, updates the initial condition-vector and computes a linear analysis. The results of the linear analysis shown in Figure 73 indicate the system to be stable. Appropriate pitch and heave states are turned on next and another linear analysis conducted. The system eigenvalues and damping ratios still show the system to be stable and a successful dynamic simulation could be expected.

The online printer is turned on and desired time history plots are requested. The first simulation command is for .33 seconds (this is the estimated time for touchdown and flap retraction) at which time the aerodynamic data is changed to 1^{o} flaps, the elevator deflection to neutral (zero) and engine spin down to idle thrust started. The remaining slideout is then simulated and the results are shown in Figures 74 through 79. Response of the principle variables shown here indicates that the system is stable and well behaved. The final linear analysis results for eigenvalues and damping ratios also show the system to be stable. Figures 80 through 87 show the remaining time history plots of the simulation for a segment of the total simulation time.

```
TABLE,TSRES     ,2  ,6
0,500,1000,1500,2000,2500
0,.3
.4,.4,.592,.696,.76,.82
.832,.9,.922,.966,1,1.04
TABLE,TBTES,11
0,.35,.4,.5,.6,.7,.8,.9,1.,1.1,1.2
0,.08,.12,.17,.23,.31,.4,.5,.625,.775,.94
TABLE,TBPES,11
0,.35,.4,.5,.6,.7,.8,.9,1.,1.1,1.2
1.,1.19,1.26,1.415,1.66,2.,2.564,3.1,3.8,4.4,5.
TABLE,TPOES,10
1,1.02,1.06,1.13,1.2,1.26,1.38,1.65,1.78,1.78
0,24,42,60,72,78,84,90,93,120
TABLE,ABLTK,8
3.1,.8,37,1,9.33,5.6,55,1
9.44,6.09,56.9,1,7.8,3.1,63,1
TABLE,XYZTK,40
86.72,2.83,-12,78.75,83.89,7.61,-14.7,56.25
79.83,10.98,-16.9,33.75,75.08,12.95,-17.1,11.25
69.25,2.03,-10,0.,62.75,2.03,-10,0
56.5,2.03,-10,0,50.5,2.03,-10,0
44.5,2.03,-10,0,38.5,2.03,-10,0
32.5,2.03,-10,0,26.5,2.03,-10,0
20.56,2.03,-10.5,0,14.69,2.03,-11.08,0
8.81,2.03,-11.66,0,2.94,2.03,-12.24,0
-2.73,13.73,-19.7,-11.25,-7.61,11.39,-18.3,-33.75
-12.72,8.5,-15.3,-56.25,-12.46,2.48,-13.1,-78.75
TABLE,DSMTK,30
7,1,.2,7,1.23,.2
7,1.41,.2,7,1.42,.2
6.5,1,.2,6.5,1,.2
6,1,.8,6,1,.8
6,1,.8,6,1,.8
6,1,.8,6,1,.8
5.875,1,.8,5.875,1.055,.8
5.875,1.11,.8,5.875,1.17,.8
8.8,1,.8,8.3,.93,.8
7.7,.78,.8,7.1,.66,.8
TABLE,IALTK,40
1,.0061,12.88,10,1,.0061,16.76,10
1,.0061,19.8,10,1,.0061,19.97,10
2,.0061,29.21,10,2,.0061,29.21,10
2,0,0,0,2,0,0,0
2,0,0,0,2,0,0,0
2,0,0,0,2,0,0,0
3,0,0,0,3,0,0,0
3,0,0,0,3,0,0,0
4,0,0,0,4,0,0,0
4,0,0,0,4,0,0,0
TABLE,RELTK,2
1.94,2.77
0.,32.4
PARAMETER VALUES
UW VA=0,VW VA=0,WW VA=0,PW VA=0,QW1VA=0,RW1VA=0
ID1VA=3,VS VA=220,ALSVA=0,S  VA=76
IDGVA=6
TCOES=1
GAXES=1,GAZES=0,XC ES=-10.9
ZO ES=-.375,PAMES=14.7,TAMES=519,IFNES=1,IBLES=0,FX1ES=0
```

Figure 72 Landing Simulation Input Data

```
AK DU=6.6,AL DU=6.3,D  DU=3,TAMDU=519,HC DU=1,FC DU=5
ZO OL=-.52,ZA OL=-4.584,ZDEOL=-.2674
XO OL=-.165,XA OL=-.7385,XDEOL=-.0764
MO OL=-.007,MALOL=-.1433,MQ OL=-6.1884
MDEOL=-.72,MAIOL=90.06,C  OL=4
XPIOL=-.32,ISWOL=3
YB DL=-.8423,YP DL=-.0215,LB DL=-.1427
LP DL=-.37,LR DL=.437,LDADL=-.187
NB DL=.0974,NP DL=-.0457,NR DL=-.1545
NDADL=.0133,B  DL=19
FY1S2=0,TX1S2=0,TZ1S2=0
AMOTK=0,DMPTK=.020
PA TK=14.7,WCUTK=0,TCUTK=560
NE TK=-20,CDGTK=.9,NSTTK=4
NPTTK=10,BSTTK=157.5,WLTTK=0,CD1TK=.6
CD2TK=.2,CDATK=.9,BSCTK=130.25,WLCTK=33.6
TAUTK=.005,EPCTK=1.,VU TK=6.
IXXSG=1190,IYYSG=1810,IZZSG=2840,IXZSG=-200
THRES=740,ELEOL=-6.6,AILDL=0
INITIAL CONDITIONS
TH ES=740,PI DU=25,PT TK=16.7,VT TK=46
PC TK=14.7,VC TK=15,U  SG=220,W  SG=9
PITSG=.4,ALTSG=3.5
ERROR CONTROLS
PI DU=.001,PT TK=.001,VT TK=.001,PC TK=.001
VC TK=.001,U  SG=.001,V  SG=.001,W  SG=.001
P  SG=.001,Q  SG=.001,R  SG=.001,ROLSG=.001
PITSG=.001,YAWSG=.001,X  SG=.001,ALTSG=.001
TH ES=.001,Y  SG=.001
NO STATES
INT CONTROL
PI DU=1,PT TK=1,VT TK=1,PC TK=1,VC TK=1
LINEAR ANALYSIS
PRINT CONTROL=4
STEADY STATE
XIC-X
LINEAR ANALYSIS
INT CONTROL
U  SG=1,W  SG=1,PITSG=1,ALTSG=1,X  SG=1,Q  SG=1,TH ES=1
LINEAR ANALYSIS
PRINTER PLOTS
DISPLAY1
ALTSG,VS,TIME
R18,VS,TIME
PITSG,VS,TIME
Q  SG,VS,TIME
TY2OL,VS,TIME
DISPLAY2
U  SG,VS,TIME
PT TK,VS,TIME
VT TK,VS,TIME
PC TK,VS,TIME
VC TK,VS,TIME
DISPLAY3
FZTTK,VS,TIME
FXTTK,VS,TIME
WTATK,VS,TIME
TH ES,VS,TIME
CPTTK,VS,TIME
TINC=.01,TMAX=.33,PRATE=5,INT MODE=2
```

Figure 72 Landing Simulation Input Data (Continued)

```
SIMULATE
XIC-X
LINEAR ANALYSIS
PARAMETER VALUES
ZO OL=-.145,ZA OL=-4.04
XO OL=-.09,XA OL=-.26
MO OL=.056,MALUL=-.2195
ELEOL=0,THRES=300
INITIAL TIME=.33,TMAX=12
SIMULATE
XIC-X
LINEAR ANALYSIS
```

Figure 72 Landing Simulation Input Data (Concluded)

/*/*/*/ LINEAR ANALYSIS /*/*/*/

STATE NAME	OPERATING POINT	PERTURBATION SIZE	INTEGRATOR CONTROL
1 TH ES	740.00	.100E-02	0
2 PI DU	25.128	.100E-02	1
3 PT TK	14.747	.100E-02	1
4 VT TK	40.010	.100E-02	1
5 PC TK	14.700	.100E-02	1
6 VC TK	14.561	.100E-02	1
7 U SG	220.00	.100E-02	0
8 V SG	0.	.100E-02	0
9 W SG	5.0000	.100E-02	0
10 P SG	0.	.100E-02	0
11 Q SG	0.	.100E-02	0
12 R SG	0.	.100E-02	0
13 ROLSG	0.	.100E-02	0
14 PITSG	.40000	.100E-02	0
15 YAWSG	0.	.100E-02	0
16 X SG	0.	.100E-02	0
17 Y SG	0.	.100E-02	0
18 ALTSG	3.5000	.100E-02	0

RATES AT OPERATING POINT

1 TH ES = 0.	2 PI DU = .22211E-06	3 PT TK = .75719E-06	4 VT TK = -.14038E-06	5 PC TK = -.20491E-05
6 VC TK = .33931E-05	7 U SG = -1.6695	8 V SG = 0.	9 W SG = -.72895	10 P SG = 0.
11 Q SG = .60746E-01	12 R SG = 0.	13 ROLSG = 0.	14 PITSG = 0.	15 YAWSG = 0.
16 X SG = 220.06	17 Y SG = 0.	18 ALTSG = -7.4639		

STABILITY MATRIX

	PI DU	PT TK	VT TK	PC TK	VC TK
PI DU	-31.32	16.19	0.	0.	0.
PT TK	.6441	-57.05	87.34	42.94	0.
VT TK	0.	.1212E-01	-200.0	-97.61	0.
PC TK	-.3501E-04	1.245	0.	-.1673E+05	235.8
VC TK	0.	.2525	0.	2359.	-200.0

5 EIGENVALUES

	REAL	IMAGINARY	NATURAL FREQ.	DAMPING RATIO
1	-30.7783	0.	30.7783	1.00000
2	-57.5619	0.	57.5619	1.00000
3	-166.417	0.	166.417	1.00000
4	-200.000	0.	200.000	1.00000
5	-16764.9	0.	16764.9	1.00000

Figure 73 Landing Approach Condition Linear Analysis

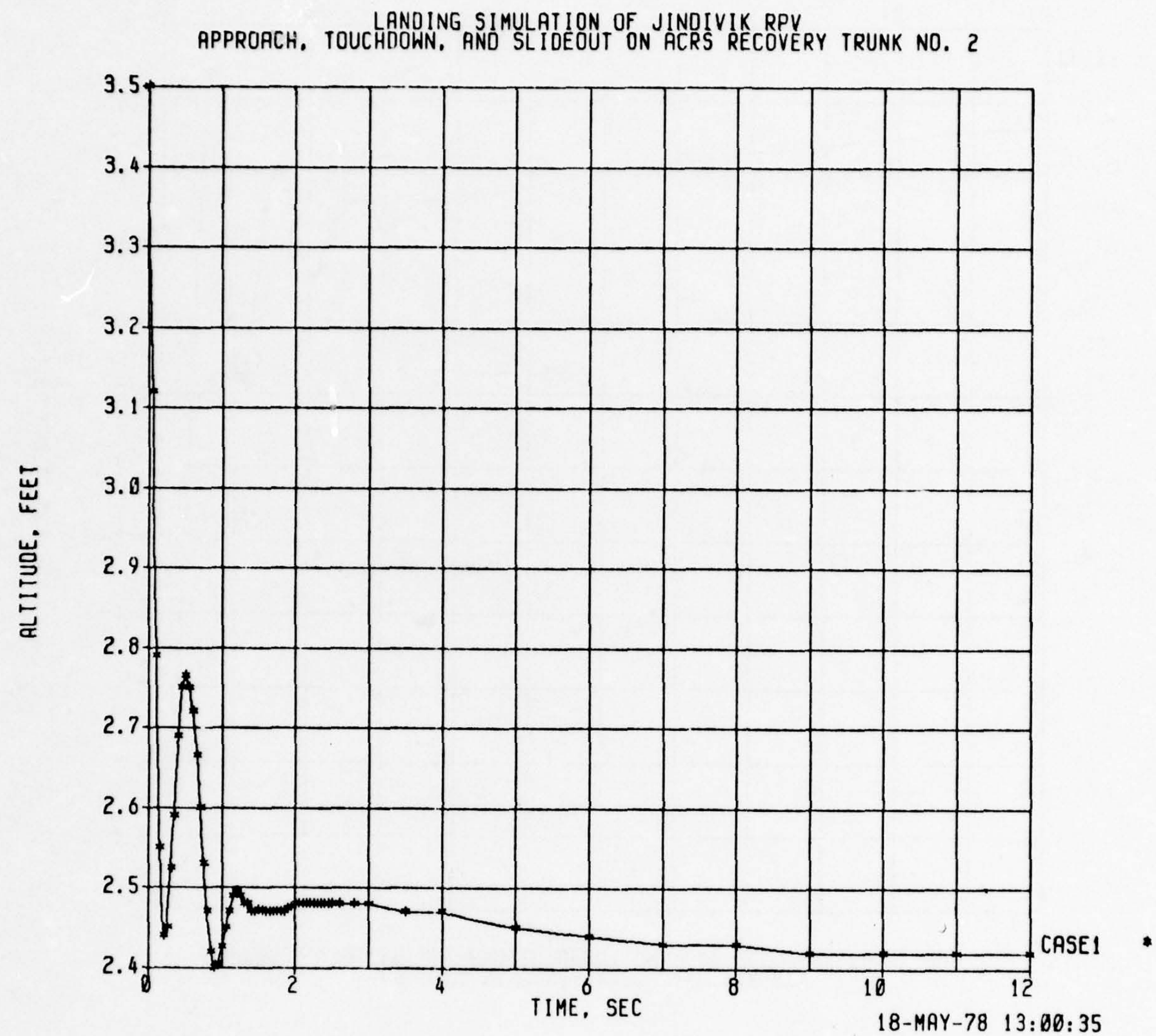


Figure 74 Time history Plot of Landing Simulation, Altitude

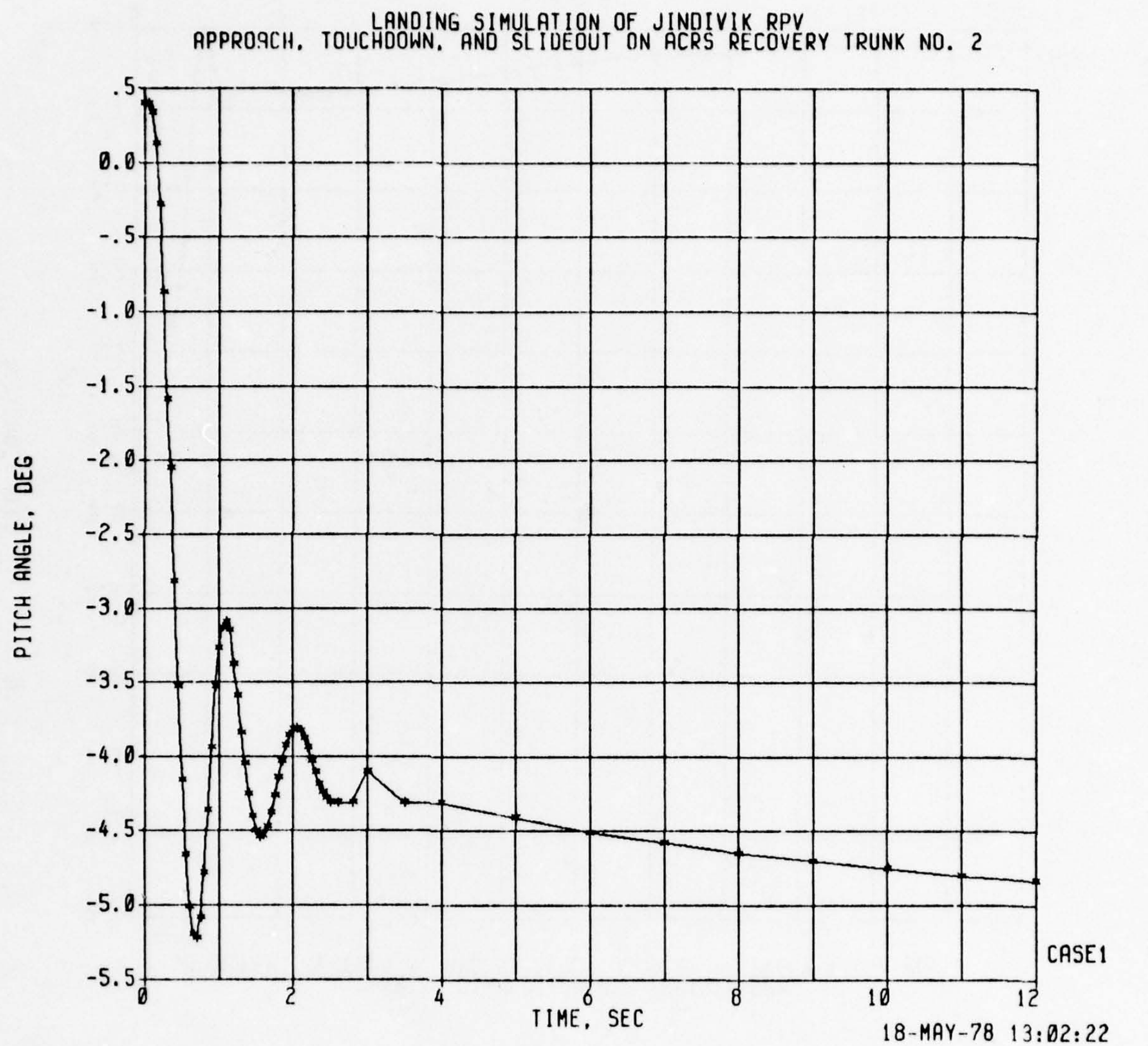


Figure 75 Time History Plot of Landing Simulation, Pitch Angle

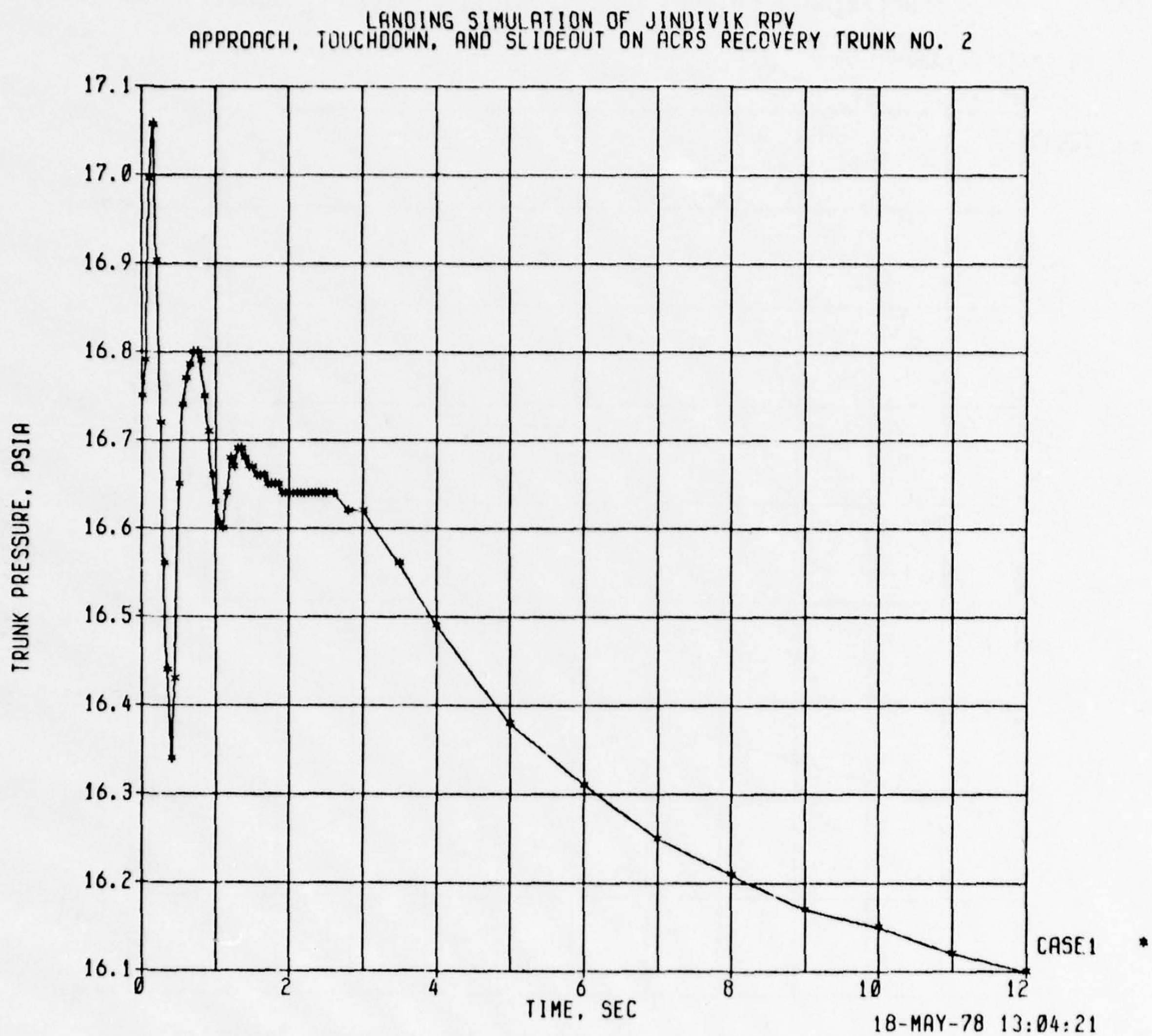


Figure 76 Time History Plot of Landing Simulation, Trunk Pr.

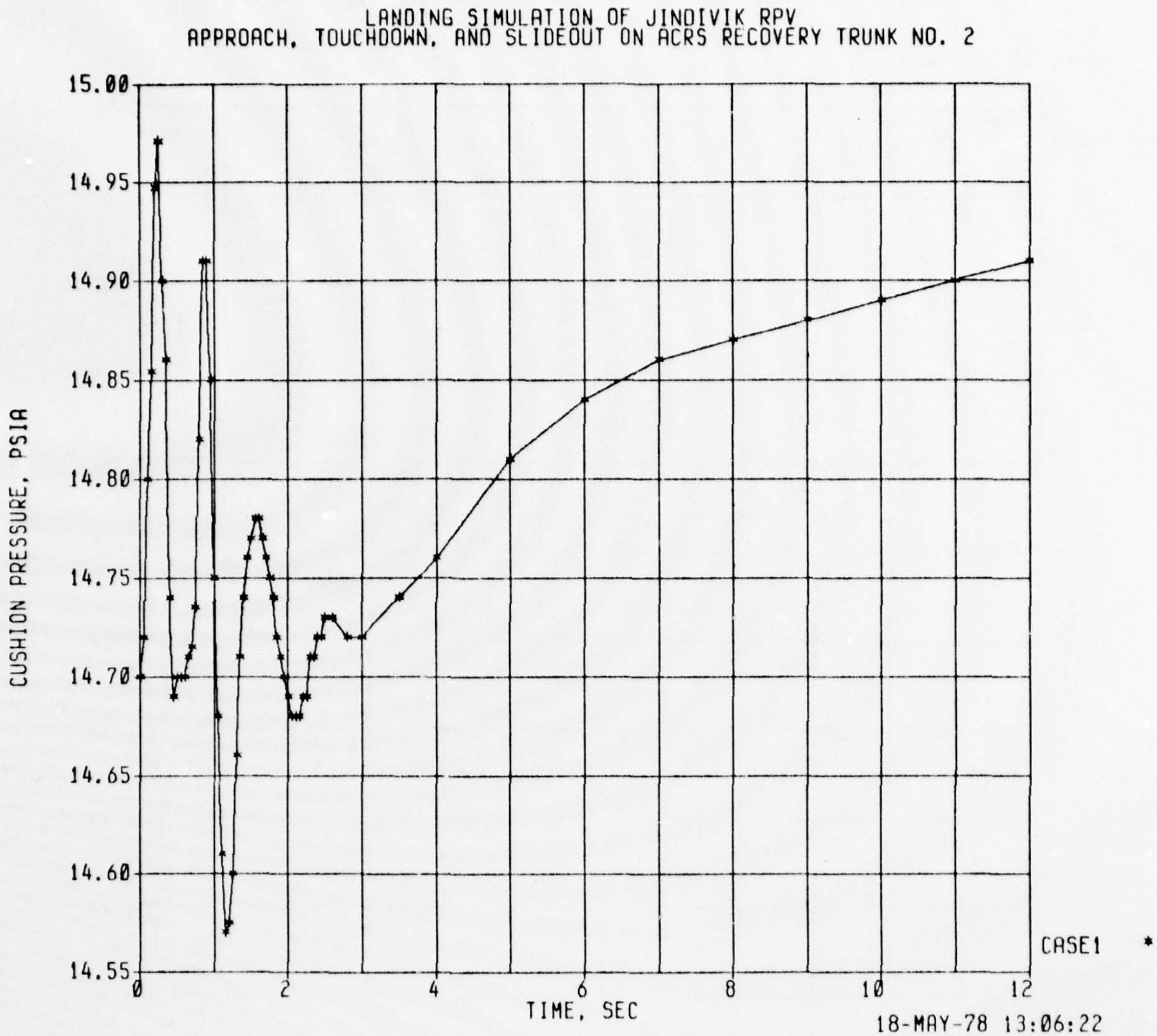


Figure 77 Time History Plot of Landing Simulation, Cushion Pr.

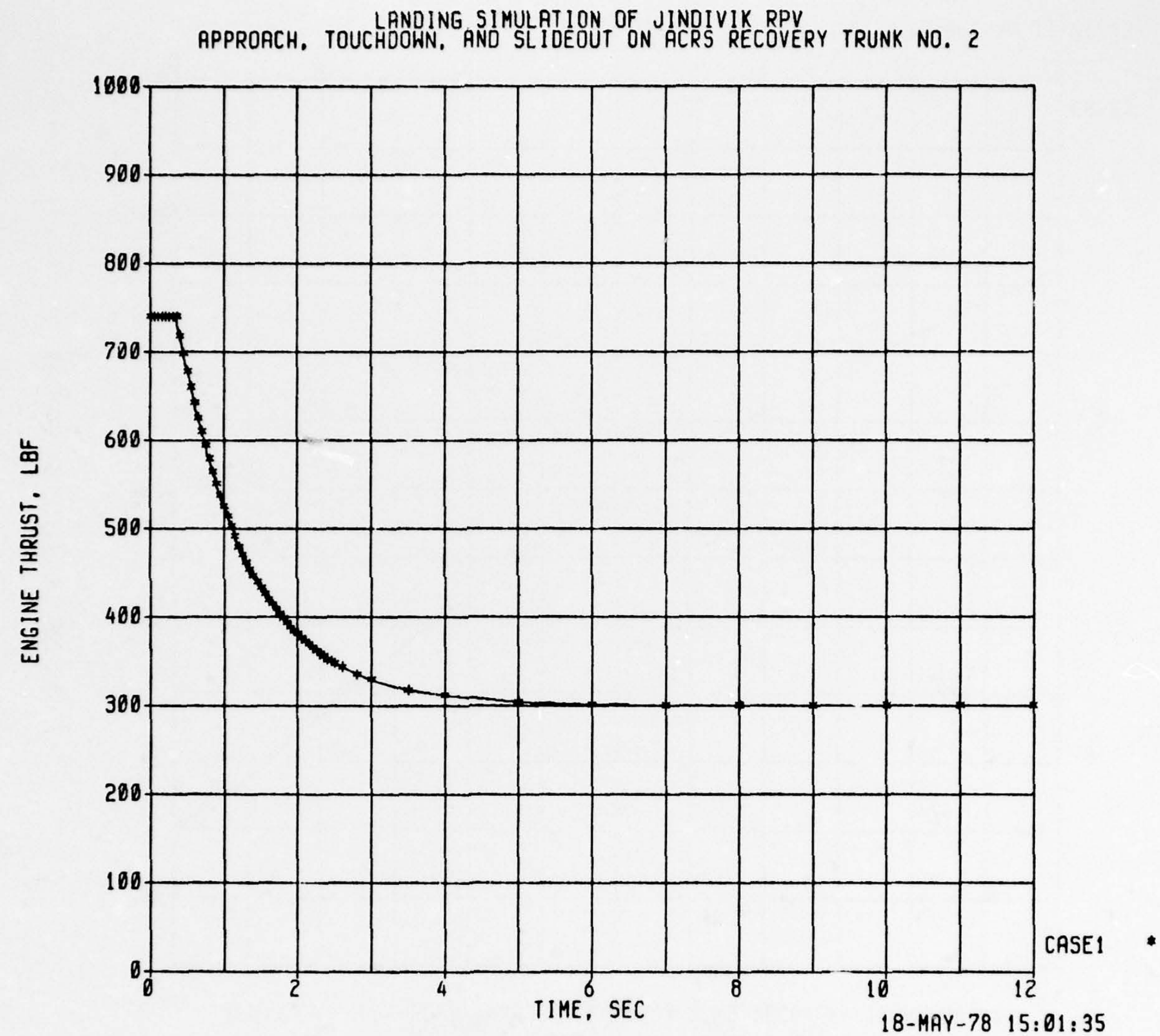


Figure 78 Time History Plot of Landing Simulation, Engine Thrust

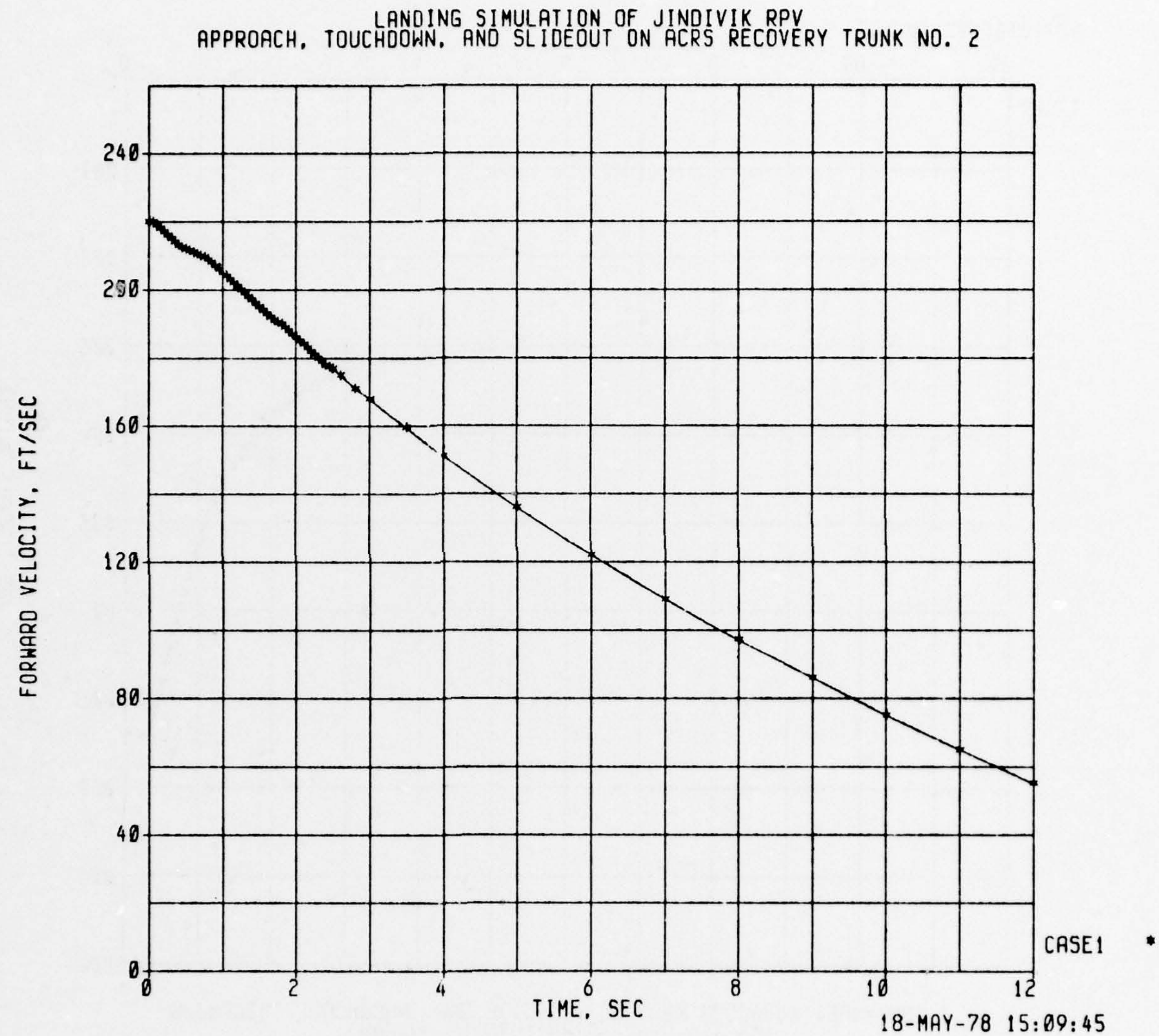


Figure 79 Time History Plot of Landing Simulation, Forward Velocity

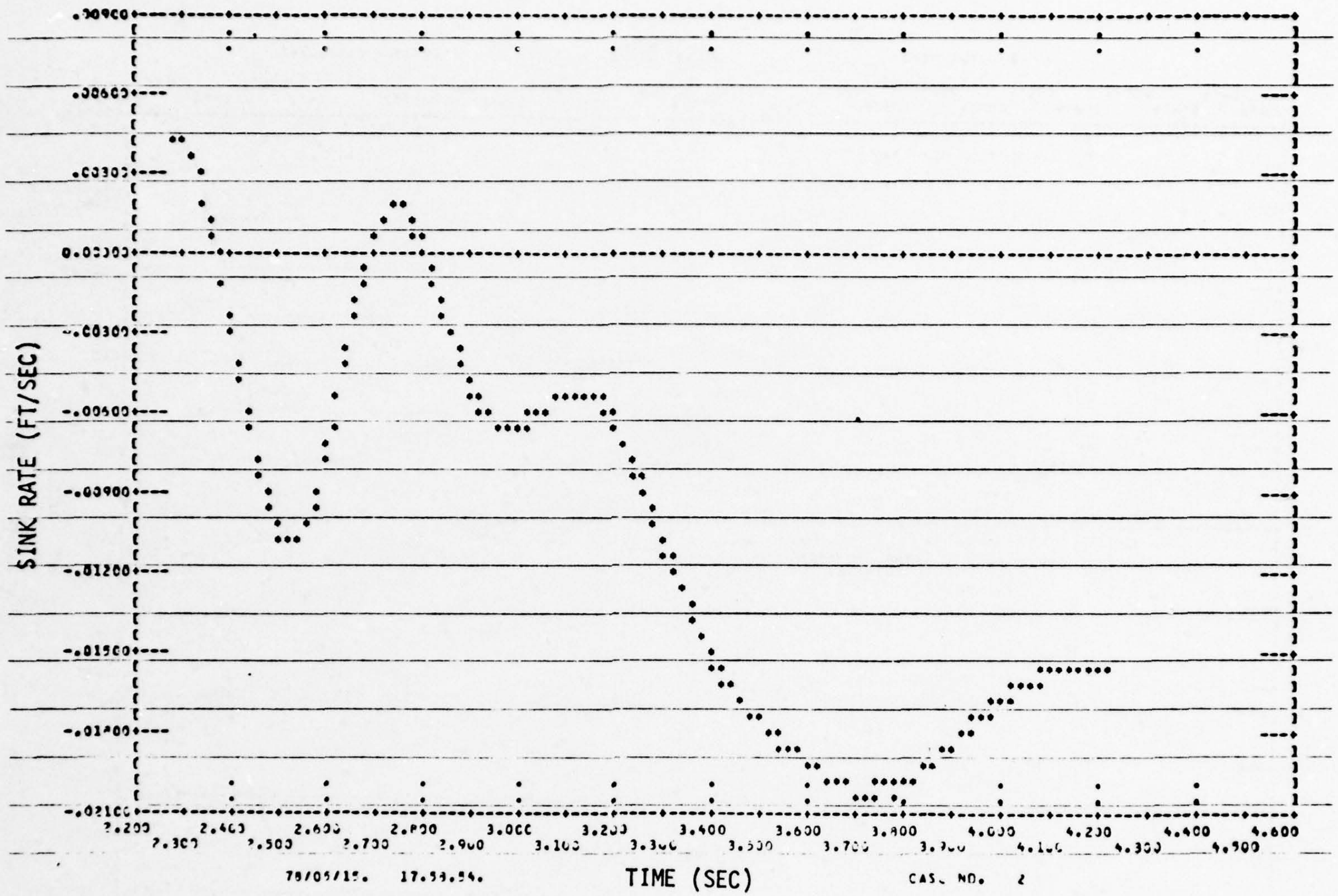


Figure 80 Time History Plot of Landing Simulation, Sink Rate

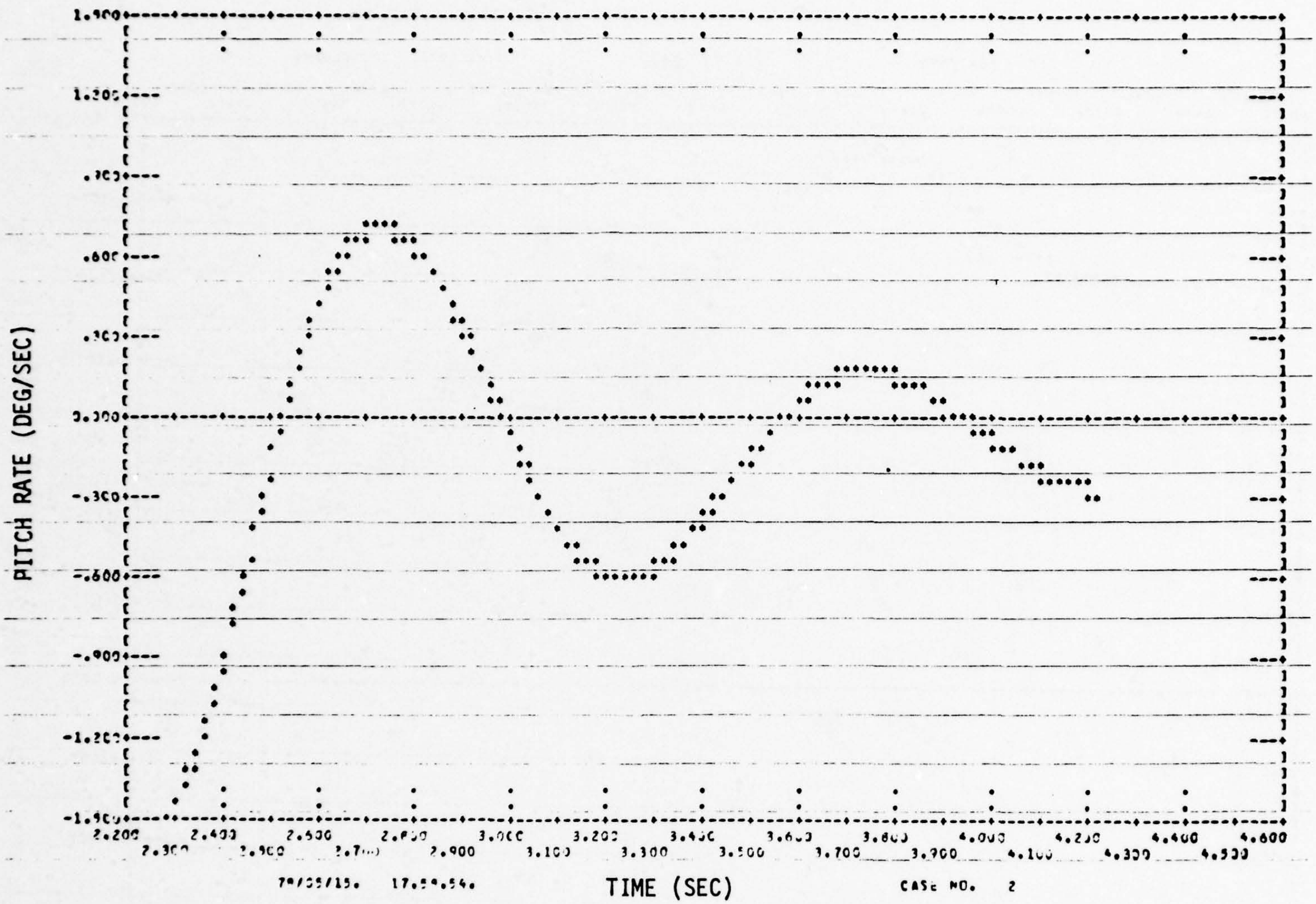


Figure 81 Time History Plot of Landing Simulation.
Pitch Rate

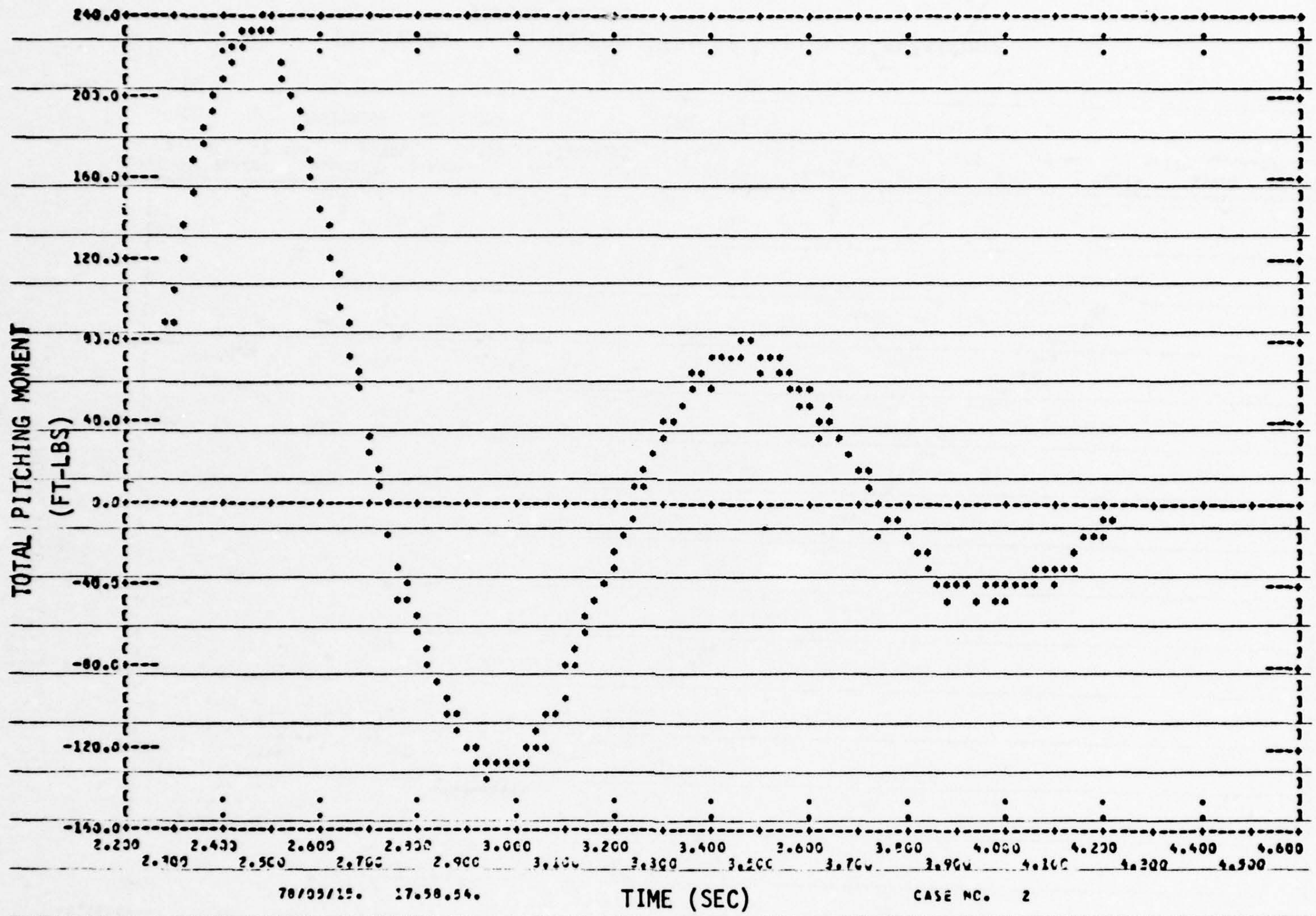


Figure 82 Time History Plot of Landing Simulation,
Total Pitch Moment

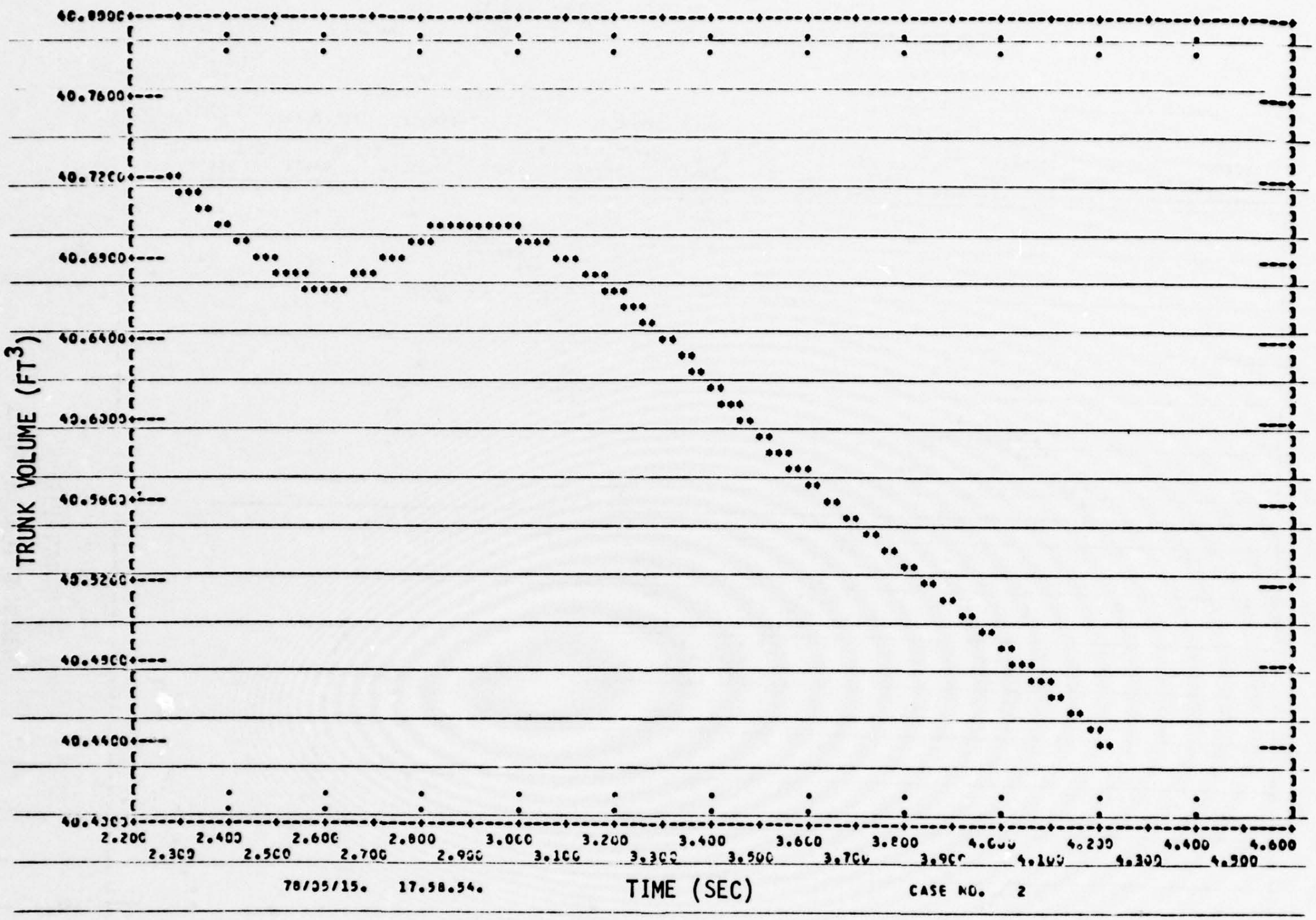


Figure 83 Time History Plot of Landing Simulation, Trunk Volume

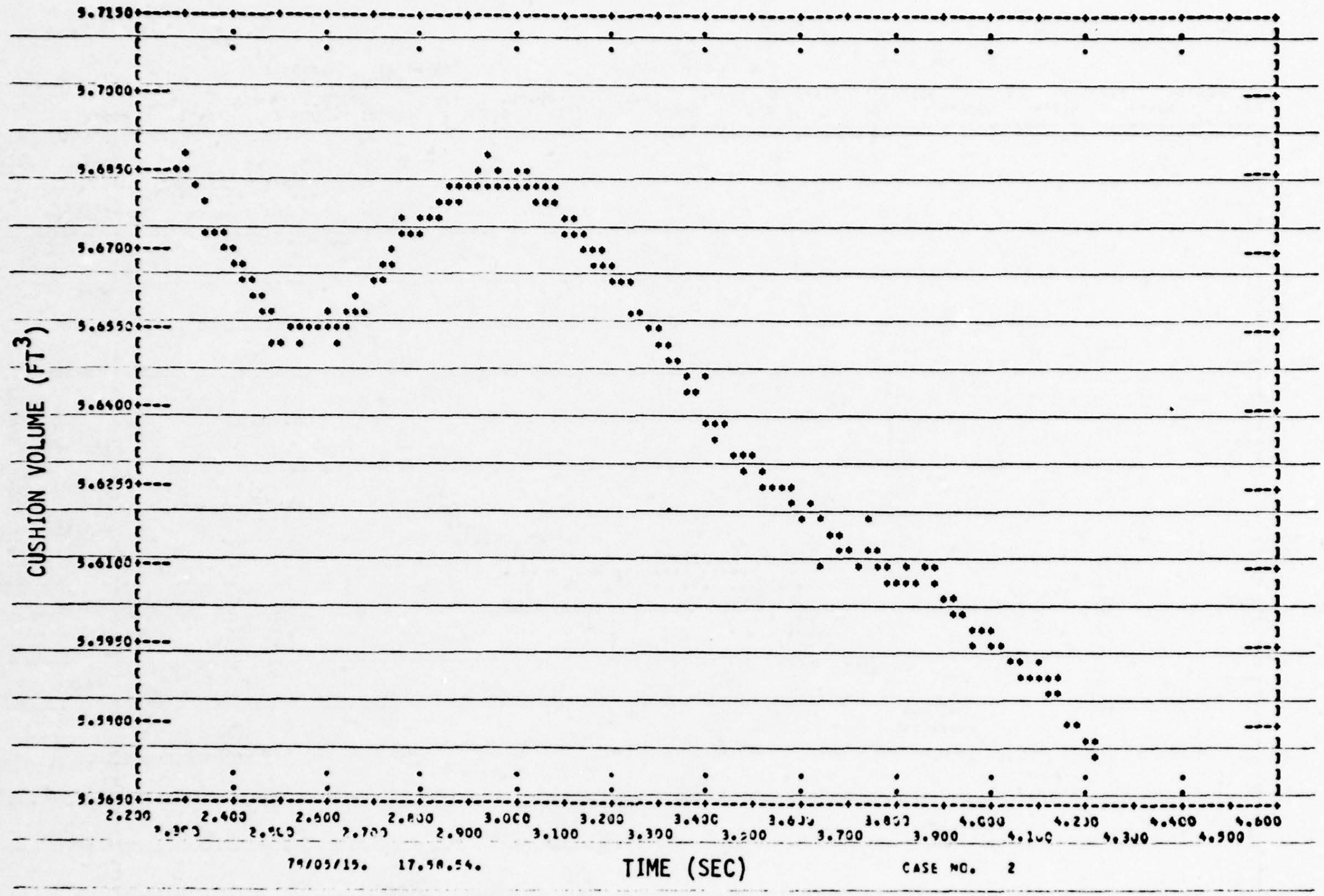


Figure 84 Time History Plot of Landing Simulation,
Cushion Volume

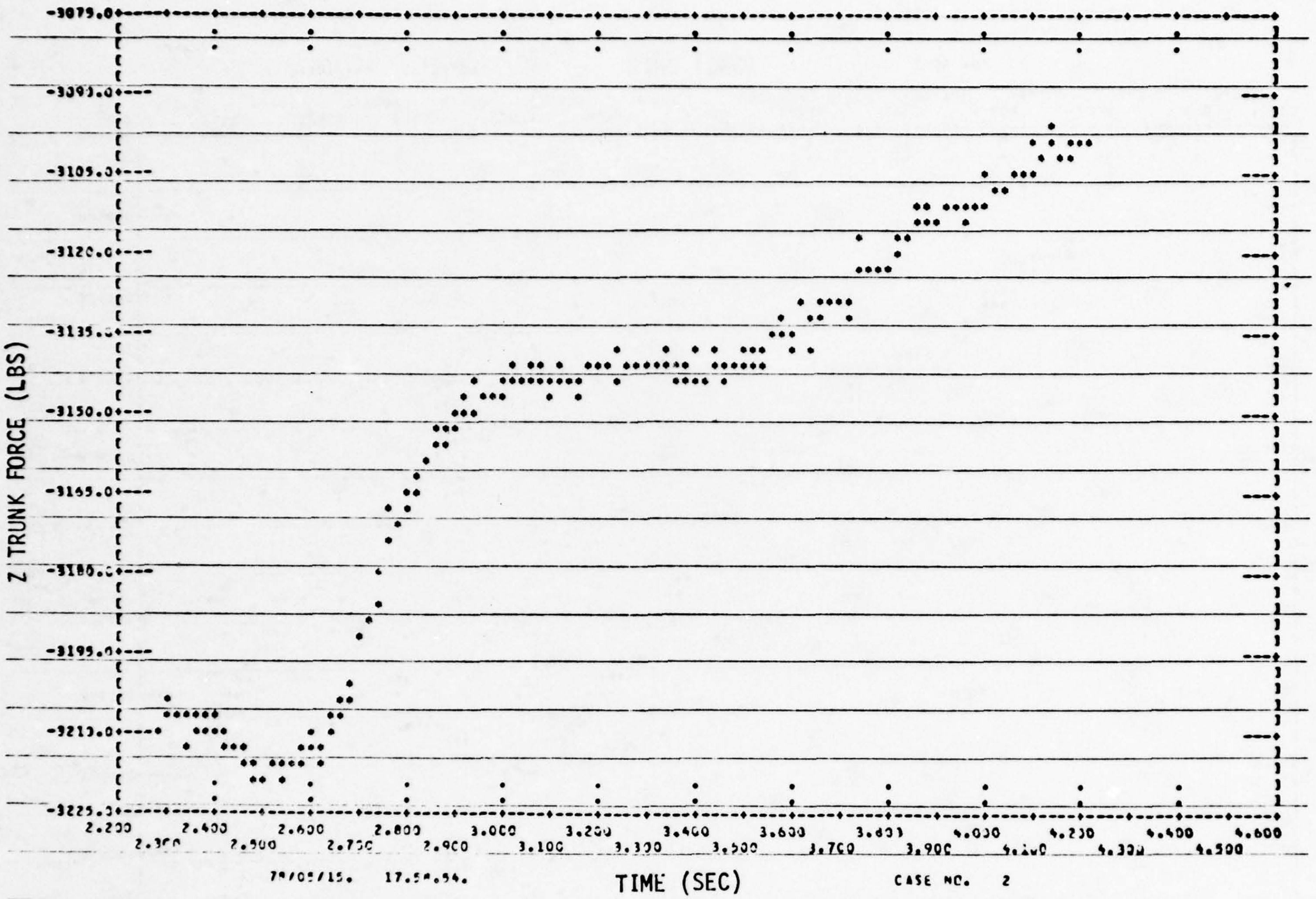


Figure 85 Time History Plot of Landing Simulation, Trunk Vertical Force

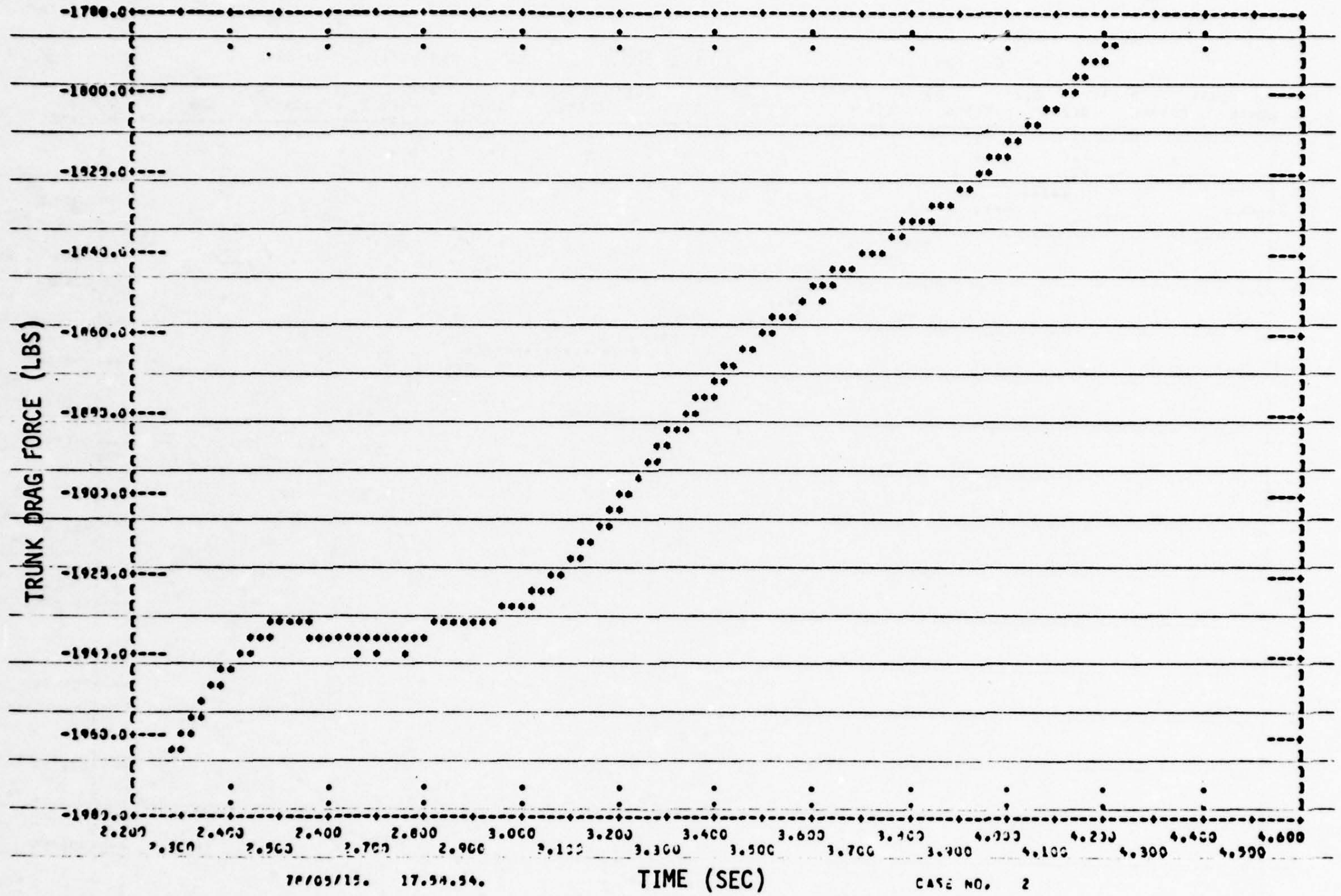


Figure 86 Time History Plot of Landing Simulation, Trunk Drag Force

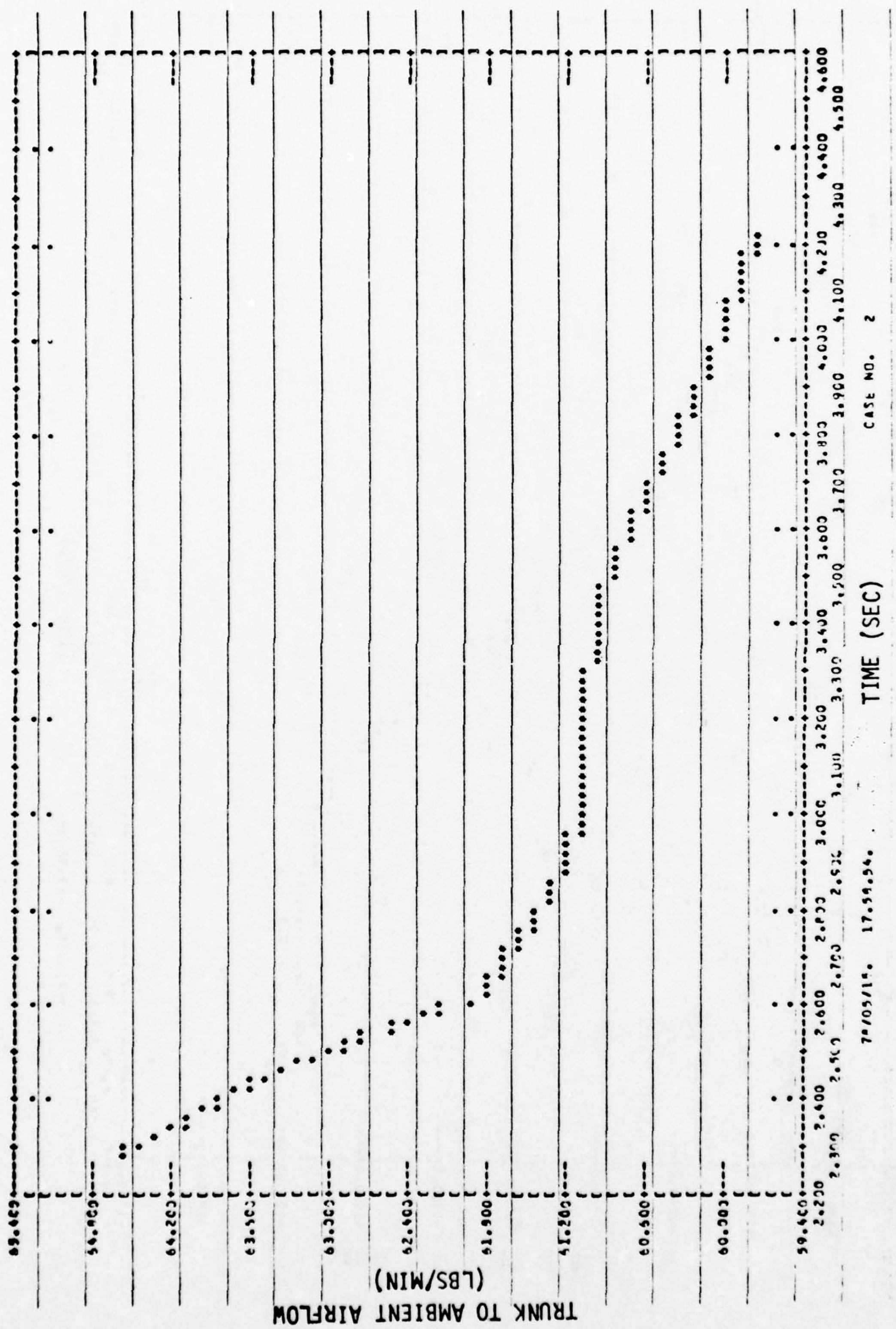


Figure 87 Time History Plot of Landing Simulation, Trunk to Ambient Airflow

SECTION V

JINDIVIK TAKEOFF SIMULATION

5.1 Objectives

The objective of this Task 5 simulation was:

- o Using components developed in Tasks 1 and 3 simulate the take-off run, rotation and climbout for the Jindivik RPV.

5.2 Technical Approach

The 6 DOF rigid body model was used along with other components developed in Tasks 1 and 3 to simulate the take off run, rotation and climbout for the Jindivik RPV. The ACRS #2 trunk geometry data was used for the simulation with the ACTS #1 lubrication hole pattern. The trunk relief valve was eliminated. 19^{o} flaps were used for the entire take off simulation. The model is basically the same as for the landing except that the ACLS turbofan module is added and the steady state (trim) calculation is obtained via STEADY STATE command and not by using optimal controller. A variable elevator deflection schedule was used to avoid severe pitch-down or pitch-up of the vehicle. The assumptions made for the analysis are as follows:

Aircraft weight = 3200 lbs.

I_{xx} = 2414 slug ft^2

I_{yy} = 1840 slug ft^2

I_{zz} = 4082 slug ft^2

I_{xz} = -200 slug ft^2

Rotation speed = 130 knots

Engine Thrust = 2500 lbs.

Elevator for Rollout = -5 to -8^{o}

Elevator at Rotation = -12^{o}

ACLS fan ON

Coefficient of sliding friction = .2

Flaps for Rollout = 19^o

Damping coefficient for trunk = .02 lb-sec/in/in^2

Aerodynamic data used for 19^o flaps was the same as for the landing simulation, Table 6. The ACLS fan performance maps used are shown in Figures 88 and 89 and were extracted from Reference 5. The conversion of the fan map, Figure 89, into a two dimensional table input format for EASY usage is explained in detail in the User's Manual Reference 1, Section 6.4.

5.3 Simulation Description and Results

The modules used to simulate the take-off run and rotation dynamics are the basic airplane components SG, VA, OL, and DL, the engine component ES, the airflow system components DU and FT, the trunk component TK, and a summing junction S2. Figure 90 shows the model description for the take-off simulation. The purpose of these components was explained in the landing simulation (Section 4.3). The fan component FT which is the only new component is used to augment the air inflow to the trunk. Figure 91 shows the EASY generated model schematic.

The input data used to simulate this model are shown in Figure 92. Various engine data tables and the trunk element definition arrays are all identical to those used for the landing simulation except as follows:

> The fan performance maps, Figure 88 and 89 are input respectively as Tables WC FT and TOTFT.
>
> The trunk element definition array IALTK defining the lubrication hole pattern has been revised to reflect the ACTS #1 trunk configuration.
>
> The relief valve table RELTK has been revised to zero outflow area. Note that the relief valve table must be input even with zero flow areas (i.e. where no relief valves are used), since the trunk module has a built in relief valve calculation and the missing table input will cause usage of default values of 1.99999.

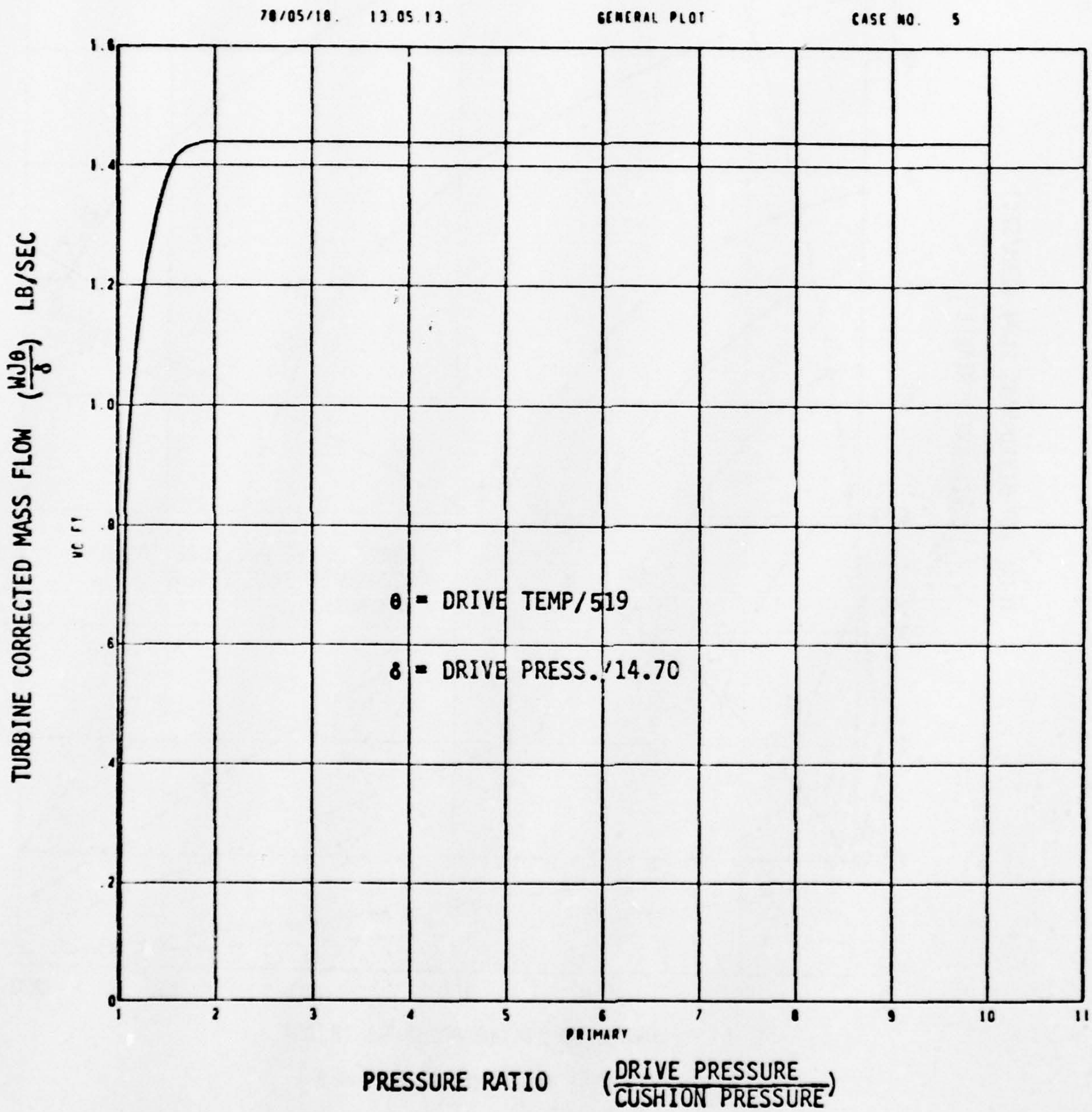


Figure 88 ACLS Fan Turbine Flow Characteristics

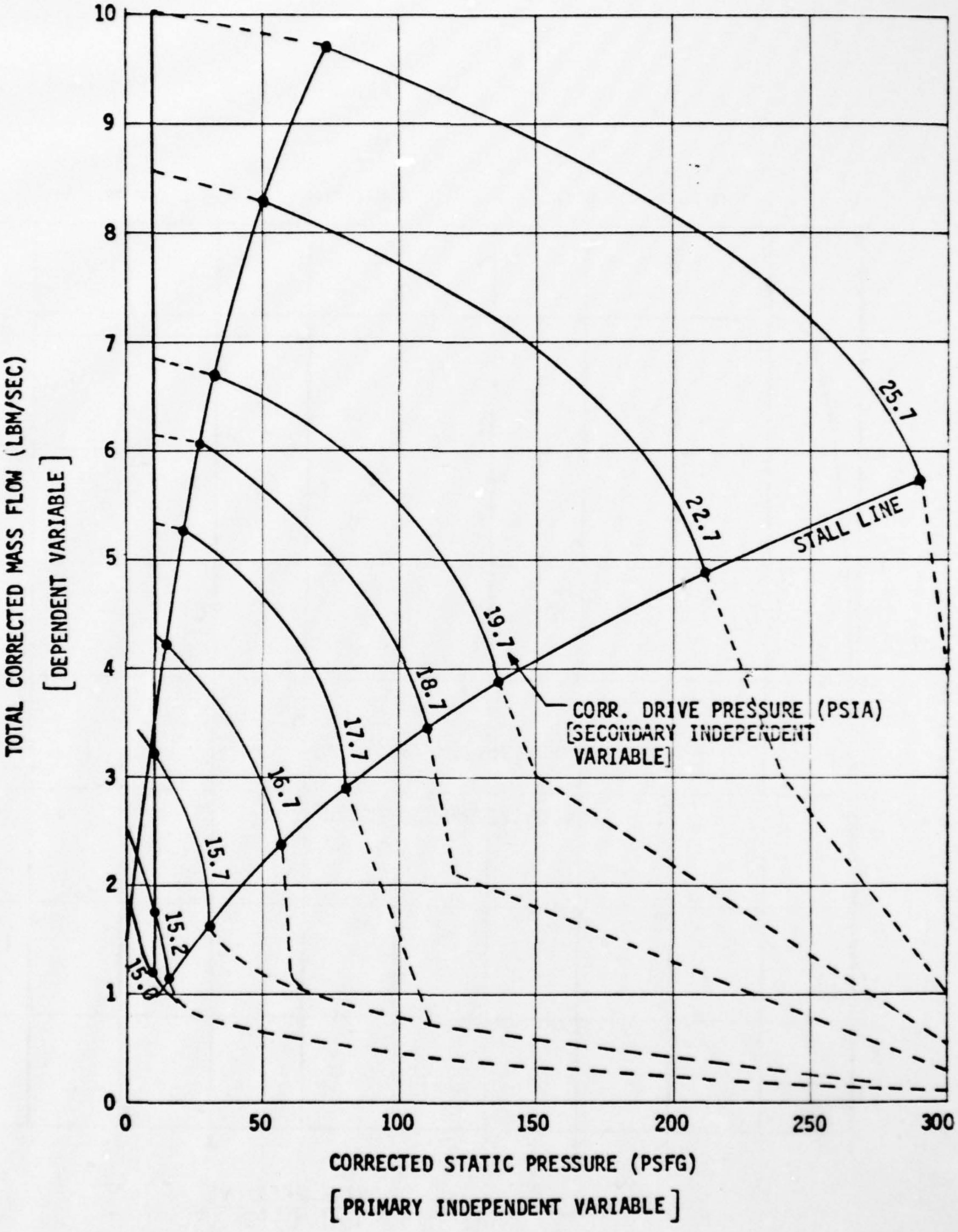


Figure 89 ACLS Fan Performance Map

```
LIST STANDARD COMPONENTS
MODEL DESCRIPTION     JINDVIK TAKEOFF TEST CASE
LOCATION=68     VA       INPUTS=SG
FORTRAN STATEMENTS
      IF(VT VA.LT.90.)ELEOL=-5
      IF(VT VA.GE.90..AND.VT VA.LT.220.)ELEOL=-8
      IF(VT VA.GE.220.)ELEOL=-12
LOCATION=74     ES       INPUTS=VA(MAC=AMN)
LOCATION=72     DU       INPUTS=ES
LOCATION=52     FT       INPUTS=DU,TK(PT=P,3)
LOCATION=12     TK       INPUTS=SG,FT(W,3=WTR,T,3=TTR)
LOCATION=34     S2       INPUTS=ES(FX=FX,1,FZ=FZ,1,TY=TY,1)
TK(FXT=FX,2,FYT=FY,2,FZT=FZ,2,TXT=TX,2,TYT=TY,2,TZT=TZ,2)
LOCATION= 5     OL       INPUTS=VA,S2(3,1)
LOCATION=38     DL       INPUTS=OL,VA,S2(3,1)
LOCATION=10     SG       INPUTS=OL,DL
END OF MODEL
PRINT
```

Figure 90 Model Description for Take-off Climbout

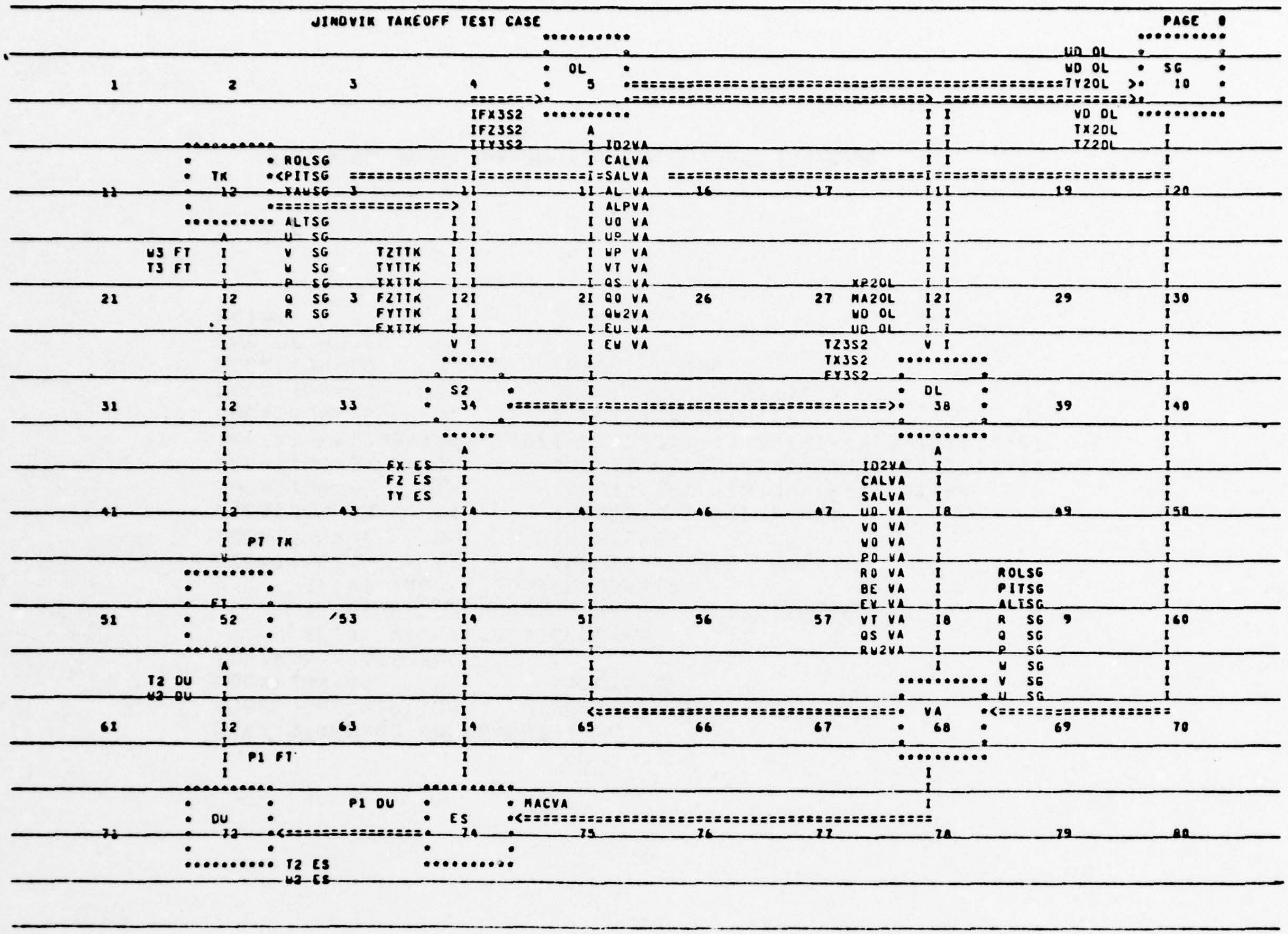


Figure 91 Jindivik Takeoff Simulation Model

```
TABLE,TSRES     ,2  ,6
0,500,1000,1500,2000,2500
0,.3
.4,.4,.592,.696,.76,.82
.832,.9,.922,.966,1.0,1.04
TABLE,TBTES,11
0,.35,.4,.5,.6,.7,.8,.9,1.,1.1,1.2
0,.08,.12,.17,.23,.31,.4,.5,.625,.775,.94
TABLE,TBPES,11
0,.35,.4,.5,.6,.7,.8,.9,1.,1.1,1.2
1.,1.19,1.26,1.415,1.66,2.,2.564,3.1,3.8,4.4,5.
TABLE,TPOES,10
1.0,1.02,1.06,1.13,1.2,1.26,1.38,1.65,1.78,1.78
0,24,42,60,72,78,84,90,93,120
TABLE,WC FT,13
1,1.1,1.2,1.3,1.4,1.5,1.6,1.7,1.8,1.9,2,2.5,10
0,.925,1.12,1.24,1.325,1.38,1.415,1.43,1.435,1.44,1.44,1.44,1.44
TABLE,TOTFT     ,16  ,9
15,15.2,15.8,16.6,17.7,18.7,19.7,22.7,25.7
300,270,240,210,180,150,120,110,80,70
60,50,40,30,20,10
.1,.15,.2,.25,.3,.35,.4,.45,.5,.55
.6,.65,.7,.75,.8,1.2
.1,.15,.2,.25,.3,.35,.4,.45,.5,.55
.6,.65,.7,.8,.9,1.76
.1,.2,.3,.4,.5,.6,.7,.8,.9,1.0
1.1,1.2,1.3,1.98,2.76,3.19
.1,.2,.3,.4,.5,.6,.7,.8,.9,1.0
1.1,2.64,3.32,3.74,4.09,4.29
.1,.2,.3,.4,.5,.6,.7,.8,3,3.9
4.3,4.62,4.88,5.12,5.3,5.34
.3,.6,.9,1.2,1.5,1.8,2.1,3.5,4.93,5.2
5.43,5.66,5.85,6.04,6.08,6.12
.5,1,1.5,2,2.5,3,4.9,5.2,5.9,6.1
6.28,6.45,6.63,6.69,6.77,6.85
1,2,3,4.88,6.25,6.91,7.4,7.55,7.95,8.07
8.2,8.3,8.34,8.4,8.48,8.56
4,6.63,7.4,7.95,8.4,8.8,9.17,9.3,9.63,9.75
9.77,9.81,9.85,9.91,9.99,10.04
TABLE,ABLTK,8
3.1,.8,37,1,9.33,5.6,55,1
9.44,6.09,56.9,1,7.8,3.1,63,1
TABLE,XYZTK,40
86.72,2.83,-12,78.75,83.89,7.61,-14.7,56.25
79.83,10.98,-16.9,33.75,75.08,12.95,-17.1,11.25
69.25,2.03,-10,0.,62.75,2.03,-10,0
56.5,2.03,-10,0,50.5,2.03,-10,0
44.5,2.03,-10,0,38.5,2.03,-10,0
32.5,2.03,-10,0,26.5,2.03,-10,0
20.56,2.03,-10.5,0,14.69,2.03,-11.08,0
8.81,2.03,-11.66,0,2.94,2.03,-12.24,0
-2.73,13.73,-19.7,-11.25,-7.61,11.39,-18.3,-33.75
-12.72,8.5,-15.3,-56.25,-12.46,2.48,-13.1,-78.75
TABLE,CSMTK,30
7,1,.2,7,1.23,.2
7,1.41,.2,7,1.42,.2
6.5,1,.2,6.5,1,.2
6,1,.2,6,1,.2
6,1,.2,6,1,.2
6,1,.2,6,1,.2
```

Figure 92 Takeoff Simulation Input Data

```
5.875,1,.2,5.875,1.055,.2
5.875,1.11,.2,5.875,1.17,.2
8.8,1,.2,8.3,.93,.2
7.7,.78,.2,7.1,.66,.2
TABLE,IALTK,40
1,.03455,18.45,10,1,.03455,25.52,10
1,.03455,31.06,10,1,.03455,31.37,10
2,.03455,31.56,10,2,.03455,31.56,10
2,.03455,31.56,10,2,.03455,31.56,10
2,.03455,31.56,10,2,.03455,31.56,10
2,.03455,31.56,10,2,.03455,31.56,10
3,.03455,32.50,10,3,.03455,33.87,10
3,.03455,35.43,10,3,.03455,36.80,10
4,.03455,39.53,10,4,.03455,38.35,10
4,.03455,29.57,10,4,.03455,24.30,10
TABLE,RELTK,2
1.94,2.77
0.,0
PARAMETER VALUES
IDIVA=3,VS VA=10,ALSVA=0,S  VA=76
IDGVA=6
TCOES=1,THRES=2500
GAXES=1,GAZES=0,XC ES=-10.9
ZO ES=-.375,PAMES=14.7,TAMES=519,IFNES=1,IBLES=0,FXIES=0
AK DU=6.6,AL DU=6.3,D  DU=3,TAMDU=519,HC DU=1,FC DU=5
T2 FT=519,P2 FT=14.7,FC FT=5,VCLFT=.25
XO OL=-.165,XA OL=-.7385,XDEOL=-.0764
ZO OL=-.52,ZA OL=-4.584,ZDEOL=-.2674
MO OL=-.007,MALOL=-.1433,MQ OL=-6.1884
MADOL=-3.46
MDEOL=-.72,MAIOL=99.38,C  OL=4
XPIOL=-.5,ISWOL=3
YB DL=-.8423,YP DL=-.0215,LB DL=-.1427
LP DL=-.37,LR DL=.437,LDADL=-.187
NB DL=.0974,NP DL=-.0457,NR DL=-.1545
NDADL=.0133,B  DL=19
PARAMETER VALUES
AMOTK=0,DMPTK=.0200
PA TK=14.7
WCUTK=0,TCUTK=560
FYIS2=0,TXIS2=0,TZIS2=0
NE TK=-20,CDGTK=.9,NSTTK=4
NPTTK=10,BSTTK=157.5,WLTTK=0,CD1TK=.6
CD2TK=.2,CDATK=.9,BSCTK=130.25,WLCTK=33.6
TAUTK=.005,EPCTK=1.,VU TK=6
IXXSG=2414,IYYSG=1840,IZZSG=4082,IXZSG=-200
INITIAL CONDITIONS
TH ES=2500,P1 DU=30.138,P1 FT=25.409
PITSG=0,ALTSG=2.582,U  SG=17,W  SG=0
PT TK=15.89,VT TK=41.75,PC TK=15.4,VC TK=8.607
ERROR CONTROLS
P1 FT=.001,P1 DU=.001,TH ES=.001
PT TK=.001,VT TK=.001,PC TK=.001,VC TK=.001
U  SG=.001,V  SG=.001,W  SG=.001
P  SG=.001,Q  SG=.001,R  SG=.001
ROLSG=.001,PITSG=.001,YAWSG=.001
X  SG=.001,ALTSG=.001,Y  SG=.001
NO STATES
INT CONTROL=PT TK=1,VT TK=1,VC TK=1,PC TK=1
P1 DU=1,P1 FT=1,ALTSG=1,W  SG=1
```

Figure 92 Takeoff Simulation Input Data (Continued)

```
PITSG=1,Q  SG=1
LINEAR ANALYSIS
PRINT CONTROL=6
STEADY STATE
XIC-X
LINEAR ANALYSIS
PRINTER PLOTS,PLOT ON
PLOT ID=     M.K.WAHI        M/S 47-03
PLOT TABLES=TSRES,TBTES,TBPES,TPOES,WC FT,TCTFT,RELTK
DISPLAY1
WD OL,VS,TIME
TY2OL,VS,TIME
W3 FT,VS,TIME
AL VA,VS,TIME
DISPLAY2
ALTSG,VS,TIME
PITSG,VS,TIME
U  SG,VS,TIME
PT TK,VS,TIME
PC TK,VS,TIME
DISPLAY3
ELEOL,VS,TIME
VT VA,VS,TIME
FZ2OL,VS,TIME
FZTTK,VS,TIME
FX2OL,VS,TIME
INT CONTROL
X  SG=1,U  SG=1
LINEAR ANALYSIS
TINC=.1,TMAX=12,INT MODE=5,PRATE=2
SIMULATE
XIC-X
LINEAR ANALYSIS
```

Figure 92 Takeoff Simulation Input Data (Concluded)

The parametric data are input as usual and the non-zero INITIAL CONDITIONS definition completes the model data definition.

The commands beginning with NO STATES and ending with STEADY STATE calculate a steady state (trim) condition for the system and the results are shown in Figures 93. The eigenvalues and the damping ratios indicate the system to be stable. Both online and offline plotters are next turned on and plots of specific tables are requested. An example of an offline plotted table is shown in Figure 94. Desired time history plots are specified and additional states are turned on for start of take off roll. The simulation results are shown in Figures 95, 96 and 97. Four of the principal variables were also plotted on the PDP-11 computer-plotting system for better resolution and are included here as Figures 98 through 101.

The simulated takeoff run indicates divergent and subsequently convergent oscillations in the pitch and heave modes for the velocity range of 50 to 100 knots. The vehicle does, however, successfully lift off at 130 knots and climb out. As part of the investigation of this unstable response a takeoff simulation run was made without aerodynamic effects. The pitch and heave oscillations were convergent and had much lower amplitudes than those in the simulation with aerodynamic effects. A sample output of the cushion pressure plot is shown as Figure 102. Similarly a higher trunk damping coefficient (e.g. .2 to .3) tends to reduce the oscillation amplitudes but not the divergent nature of the oscillations.

```
/*/*/*/   STEADY STATE ANALYSIS   /*/*/*/

A MAXIMUM OF   30 ITERATIONS CAN BE USED

      TIME =  0.                                    STATES
  1 TH ES  =  2900.0       2 P1 DU  =  45.026        3 P1 FT  =  31.991        4 PT TK  =  16.199        5 VT TK  =  43.042
  6 PC TK  =  15.477       7 VC TK  =  9.0645        8 U  SG  =  17.000        9 V  SG  =  0.           10 W  SG  =  .23438
 11 P  SG  =  0.          12 Q  SG  =  0.           13 R  SG  =  0.           14 ROLSG  =  0.           15 PITSG  =  .78988
 16 YAWSG  =  0.          17 X  SG  =  0.           18 Y  SG  =  0.           19 ALTSG  =  2.7284

                                                     RATES
  1 R1     =  0.           2 R2     =  .32215E-06    3 R3     =  .32274E-06    4 R4     =  -.17899E-05   5 R5     =  .39414E-05
  6 R6     =  .26085E-04   7 R7     =  -.62813E-05   8 R8     =  24.114        9 R9     =  0.           10 R10    =  .21880E-07
 11 R11    =  0.          12 R12    =  -.83139E-06  13 R13    =  0.           14 R14    =  0.           15 R15    =  0.
 16 R16    =  0.          17 R17    =  17.000       18 R18    =  0.           19 R19    =  -.62936E-11

                                                     VARIABLES
  1 UO VA  =  17.000       2 VO VA  =  0.            3 WO VA  =  .23438        4 PO VA  =  0.            5 QO VA  =  0.
  6 RO VA  =  0.           7 ID2VA  =  3.0000        8 QW2VA  =  0.            9 RW2VA  =  0.           10 CALVA  =  1.0000
 11 SALVA  =  0.          12 AL VA  =  .78988       13 ALPVA  =  .78988       14 VT VA  =  17.002       15 BE VA  =  0.
 16 WP VA  =  .23438      17 UP VA  =  .70000       18 EU VA  =  -.44353      19 EV VA  =  0.           20 EW VA  =  32.171
 21 SIGVA  =  .99992      22 OC VA  =  .34351       23 QS VA  =  26.105       24 MACVA  =  .15229E-01   25 FX ES  =  2500.0
 26 FZ ES  =  0.          27 TY ES  =  -937.50      28 FSPES  =          -I   29 FSTES  =          -I   30 PPUFS  =  56.046
 31 TPUES  =  844.98      32 W2 ES  =  147.45       33 T2 ES  =  844.98       34 T2 DU  =  844.24       35 W2 DU  =  147.45
 36 T3 FT  =  620.92      37 W3 FT  =  470.53       38 FXTTK  =  -37.906      39 FYTTK  =  0.           40 FZTTK  =  -3182.5
 41 TXTTK  =  0.          42 TYTTK  =  938.35       43 TZTTK  =  0.           44 WTATK  =  176.79       45 WCATK  =  293.74
 46 WTCTK  =  293.74      47 CPTTK  =  8.5250       48 FX3S2  =  2462.1       49 FY3S2  =  0.           50 FZ3S2  =  -3182.5
 51 TX3S2  =  0.          52 TY3S2  =  .84680       53 TZ3S2  =  0.           54 FX2OL  =  2457.7       55 FZ2OL  =  -3197.1
 56 TY2OL  =  -.26699E-04 57 UD DL  =  24.287       58 WD CL  =  .21880E-07   59 MA2OL  =  99.380       60 XP2OL  =  -.50000
 61 FY2OL  =  0.          62 VD DL  =  0.           63 TX2DL  =  0.           64 TZ2OL  =  0.           65 PD SG  =  0.
 66 QD SG  =  -.83139E-06 67 RD SG  =  0.           68 RODSG  =  0.           69 PIDSG  =  0.           70 YADSG  =  0.
 71 XD SG  =  17.000      72 YD SG  =  0.

                                                     PARAMETERS
  1 IDIVA  =  3.0000       2 VS VA  =  10.000        3 ALSVA  =  0.            4 S  VA  =  76.000        5 UW VA  =  0.
  6 VW VA  =  0.           7 WW VA  =  0.            8 PW VA  =  0.            9 QWIVA  =  0.           10 RWIVA  =  0.
 11 IDGVA  =  6.0000      12 TCDES  =  1.0000       13 THRES  =  2500.0       14 GAXES  =  1.0000       15 GAZES  =  0.


SYSTEM EIGENVALUES AT THIS OPERATING POINT

                              10  EIGENVALUES
          REAL             IMAGINARY        NATURAL FREQ.      DAMPING RATIO
    1   -.154098        +-  5.05327            5.05562          .304805E-01
    2   -1.43099        +-  9.28064            9.39031          .152390
    3   -19.0184            0.                19.0184          1.00000
    4   -27.2253            0.                27.2253          1.00000
    5   -65.5533            0.                65.5533          1.00000
    6   -107.573            0.               107.573           1.00000
    7   -198.308            0.               198.308           1.00000
    8   -12683.8            0.             12683.8             1.00000
```

Figure 93 Jindivik Takeoff Simulation Steady State

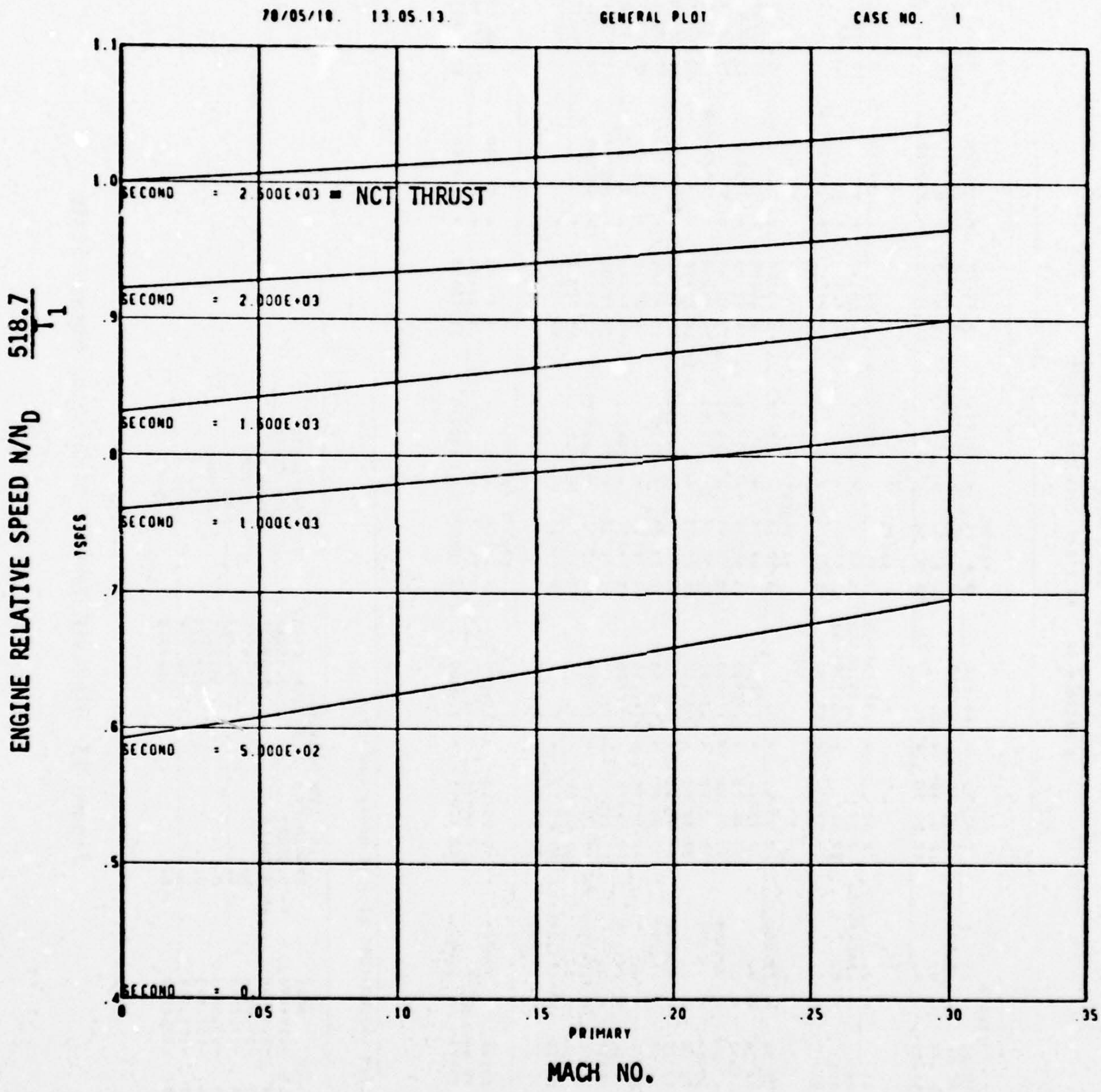


Figure 94 Tabular Input TSRES Plotted by EASY

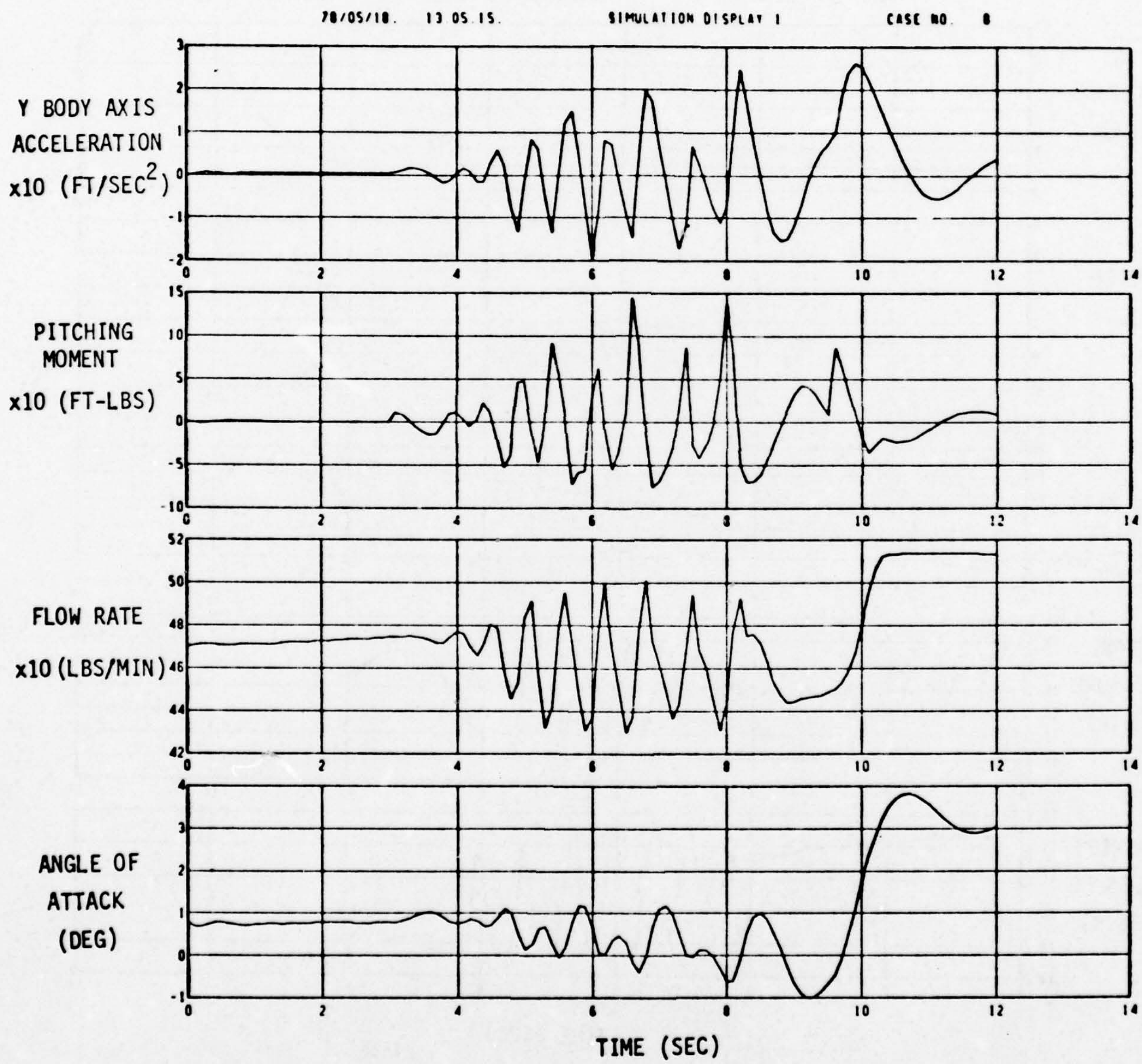


Figure 95 Takeoff Run Time History Plots, DISPLAY1

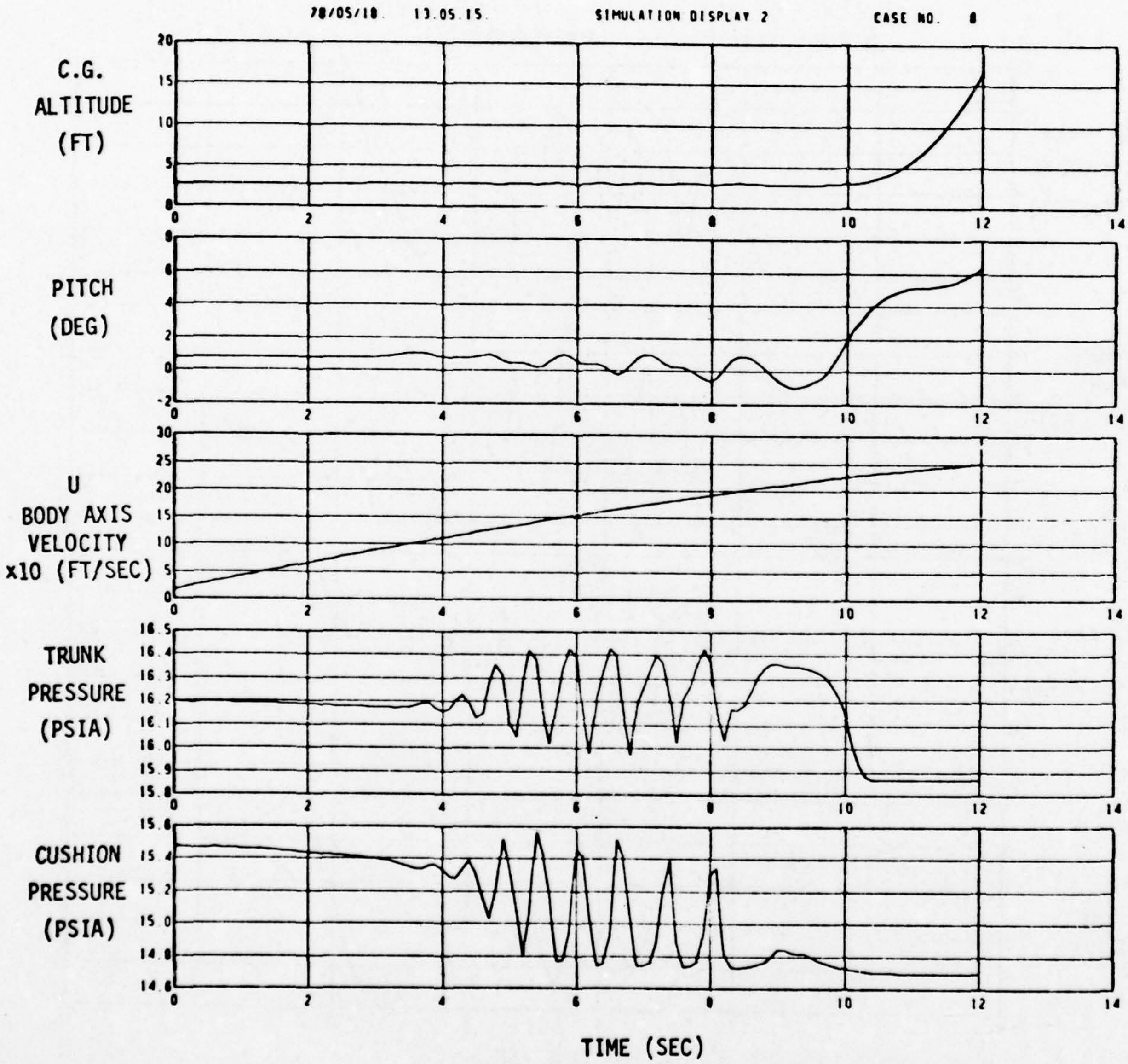


Figure 96 Takeoff Run Time History Plots DISPLAY2

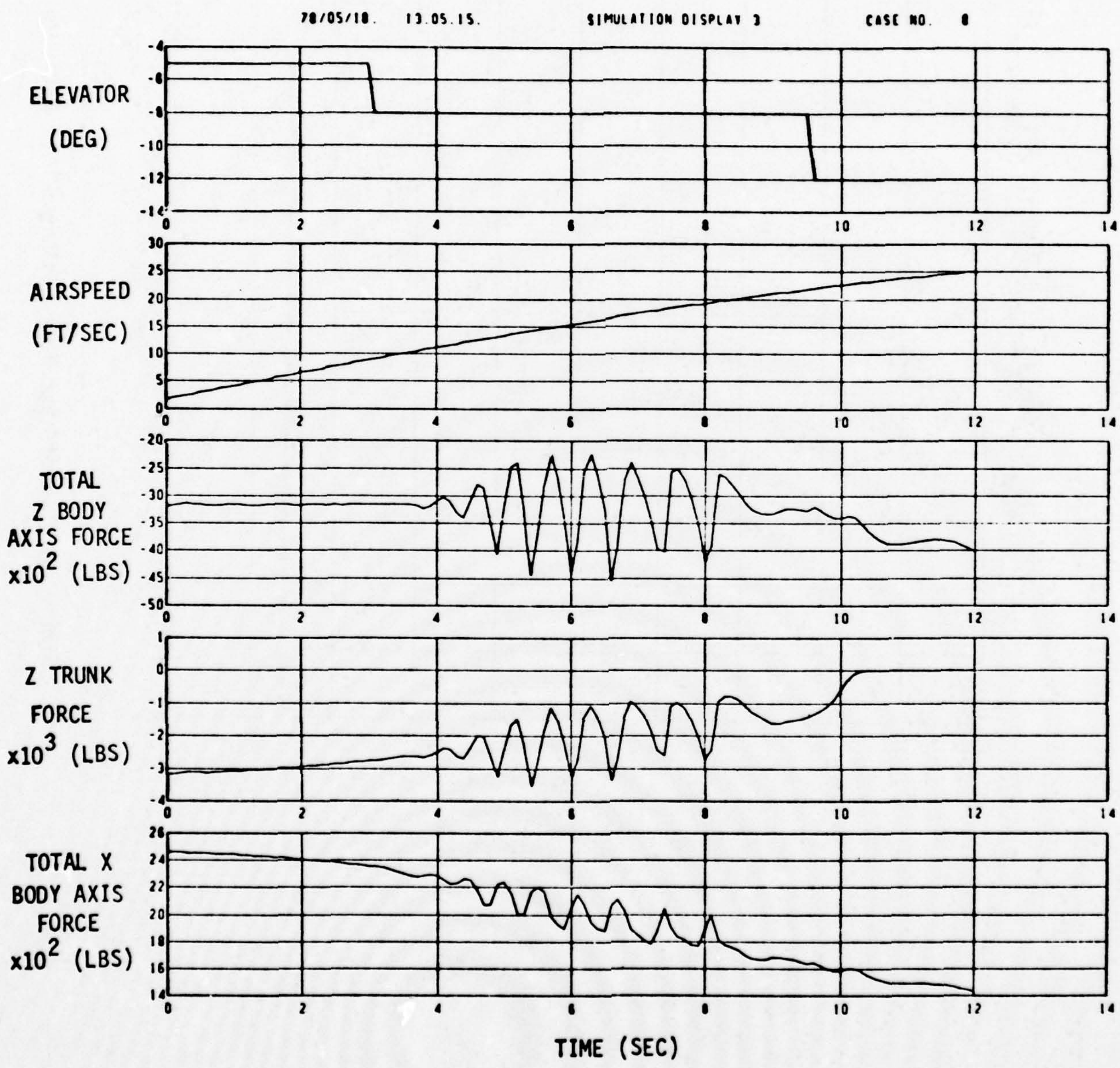


Figure 97 Takeoff Run Time History Plots, DISPLAY3

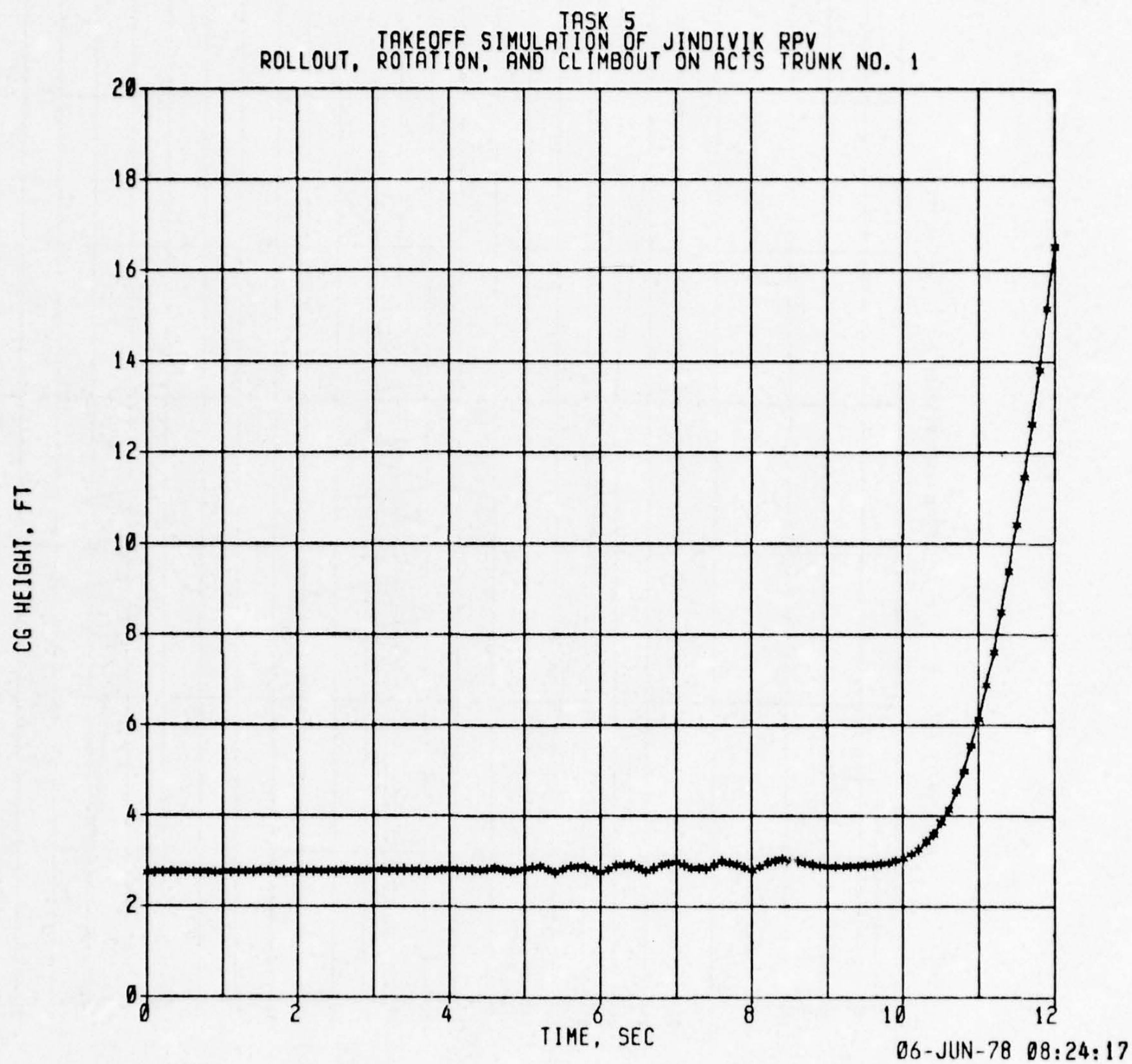


Figure 98 Takeoff Run Time History Plot, Altitude

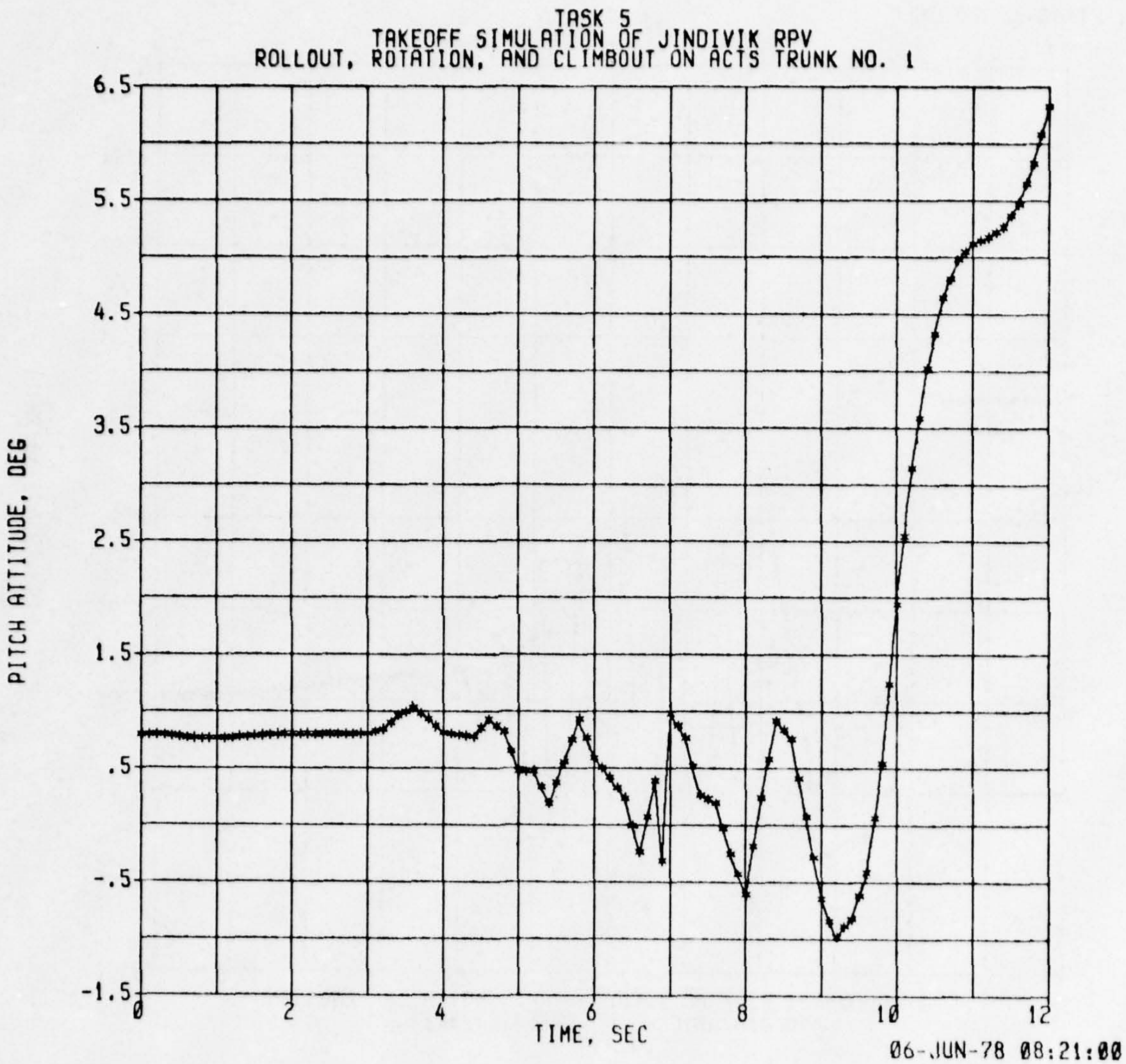


Figure 99 Takeoff Run Time History Plot, Pitch Angle

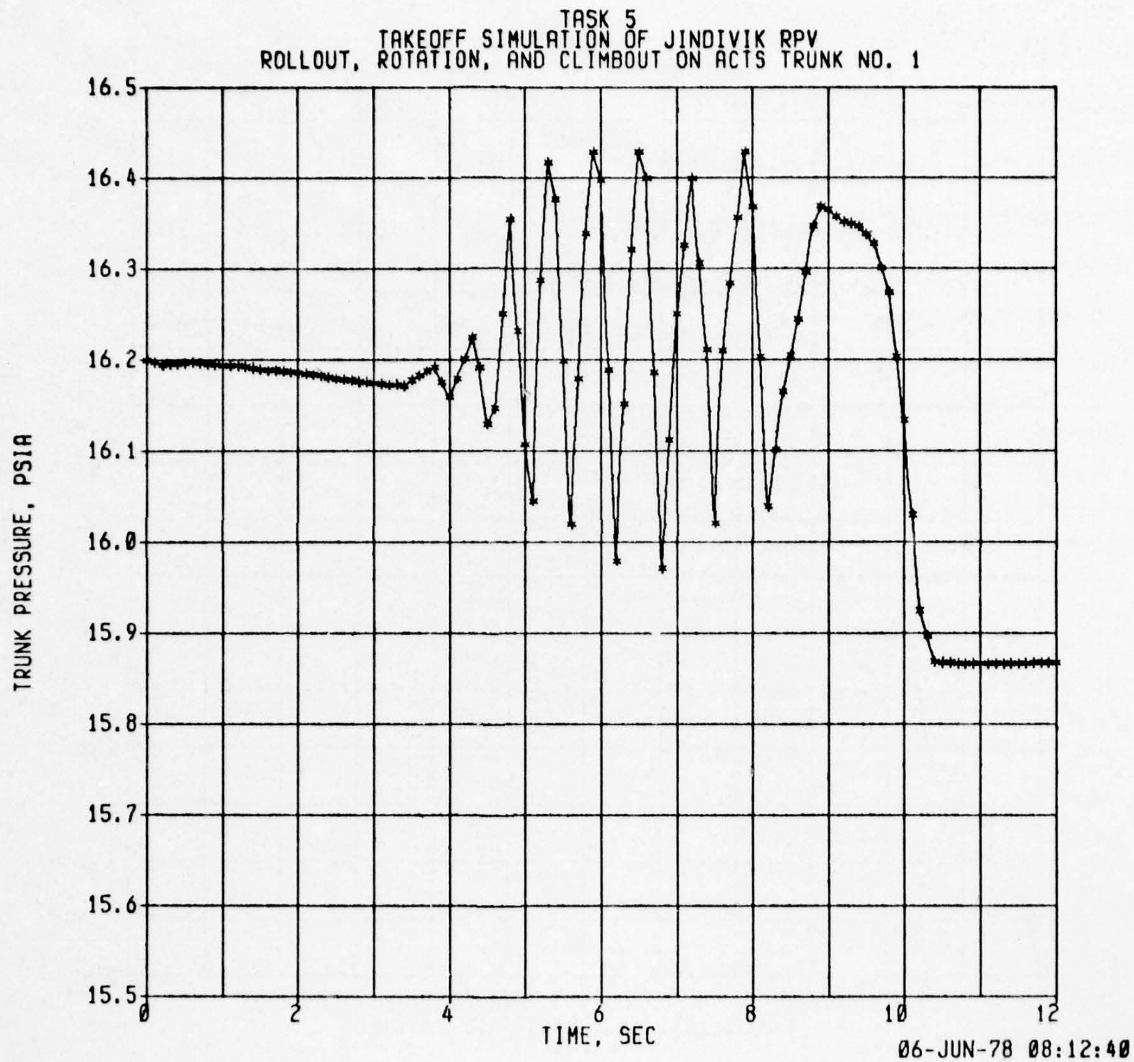


Figure 100 Takeoff Run Time History Plot, Cushion Pr.

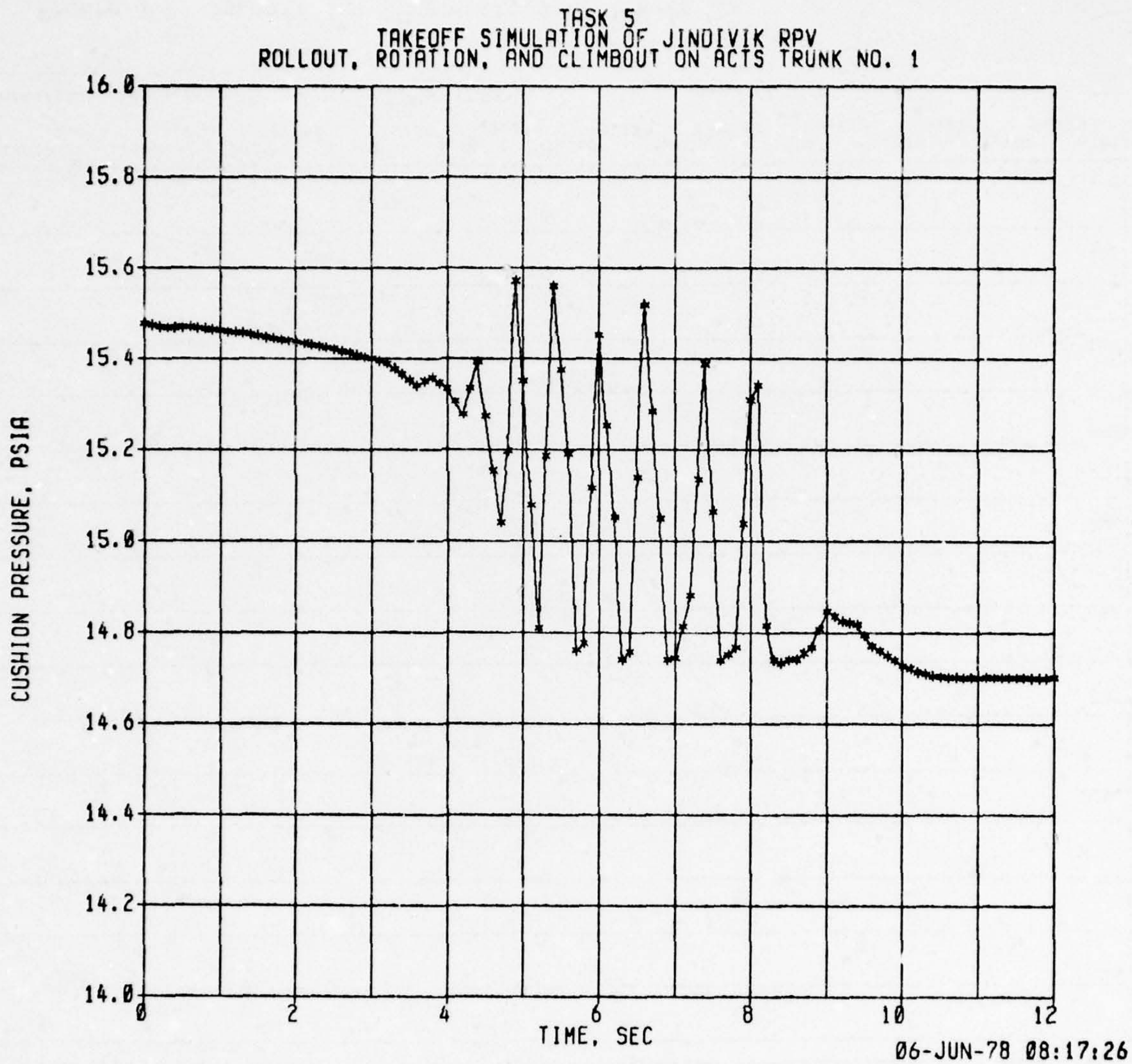


Figure 101 Takeoff Run Time History Plot, Trunk Pr.

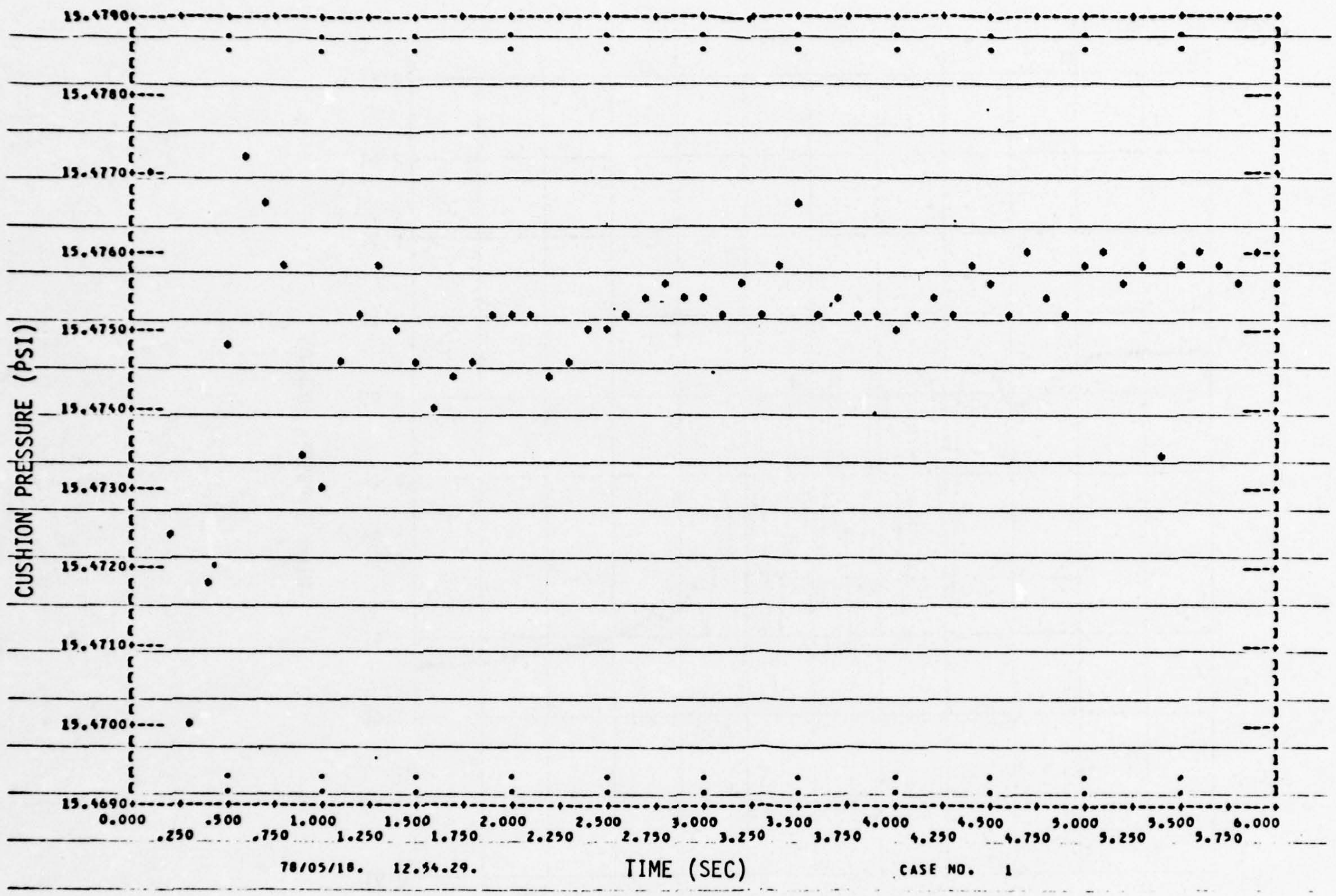


Figure 102 Takeoff Run Time History Plot With No Aerodynamics

SECTION VI

AIR BAG SIMULATION

6.1 Objectives

The objectives of the air bag simulations were to checkout and debug the air bag model component.

6.2 Technical Approach

An air bag model was developed to simulate an air bag skid system which employs two parallel inflatable membranes or bags along the underside of the fuselage, Figure 103. The basic concepts for coding the air bag skid component "AB" are common to those used for the inelastic trunk "TK". A drop test with accompanied slideout was performed to checkout the air bag component "AB" and its associated subroutines and functions.

The data used for the simulation purposes was extracted from one of the Boeing ARPV designs. Assumptions made for the analysis are as follows:

RPV Weight = 1800 lbs.

I_{xx} = 80 slug ft^2

I_{yy} = 1240 slug ft^2

I_{zz} = 900 slug ft^2

Forward speed = 100 knots

Attitude

Pitch = 5 deg

Yaw = 10 deg

Air inflow to each bag = 25 lb/min at 560^{o}R

Discharge coefficient for free portion of bag perforations = 0.6

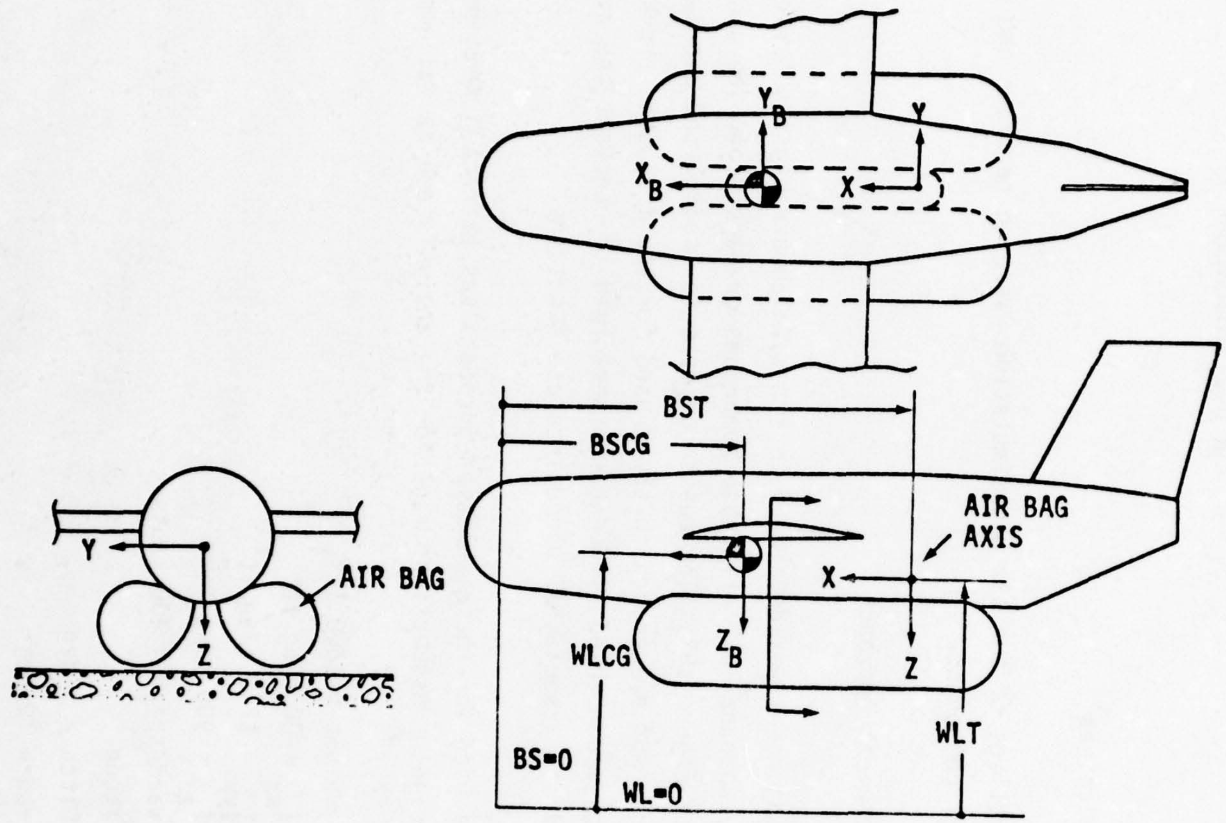


Figure 103 Basic Air Bag Geometry

Discharge coefficient for flattened portion of bag perforations = 0.2

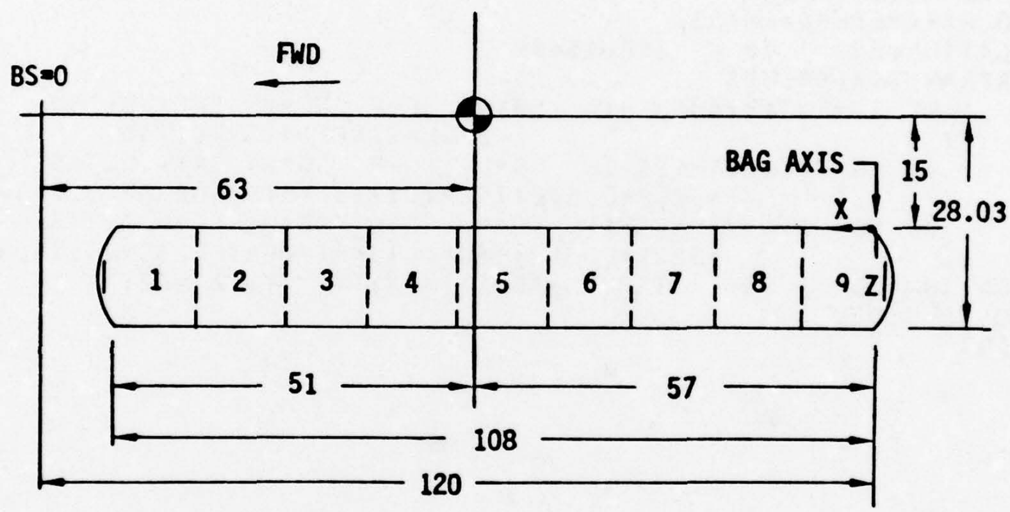
Discharge coefficient for relief valve = 0.9

Ambient air pressure = 14.7 psia

Bag damping coeffient = .02 lb-sec/in^3

Bag perforations area per unit of bag area = .0061

For input into component AB each air bag was divided into 9 segments shown below:



6.3 Simulation Description and Results

The basic EASY components required to model the vehicle and the bag dynamics are SG and AB (no aerodynamic effects are simulated). The model is identical to the trunk drop test model of Section 4.3 and a description of the model is shown here in Figure 104. The EASY generated model schematic is shown in Figure 105.

The input data used to simulate this model are shown in Figure 106. Two identical bags are used with data being input for right bag only. The bag is divided into 9 segments or elements; all of them belonging to one shape set. All elements being identical the scale factor are therefore equal to unity.

```
MODEL DESCRIPTION          COMPONENT AB CHECKOUT  (DROP)
ADD PARAMETERS=AMASS
LOCATION=65    AB    INPUTS=SG
FORTRAN STATEMENTS
      UD SG=FXTAB/AMASS-(Q  SG*W  SG-R  SG*V  SG)*.01745-
     1                           32.2*SIN(PITSG*.01745)
      VD SG=FYTAB/AMASS-(R  SG*U  SG-P  SG*W  SG)*.01745
     1             +32.2*COS(PITSG*.01745)*SIN(ROLSG*.01745)
      WD SG=FZTAB/AMASS-(P  SG*V  SG-Q  SG*U  SG)*.01745
     1             +32.2*COS(PITSG*.01745)*COS(ROLSG*.01745)
LOCATION=20    SG    INPUTS=AB(TXT=TX,TYT=TY,TZT=TZ)
END OF MODEL
PRINT
```

Figure 104 Air Bag Model Description

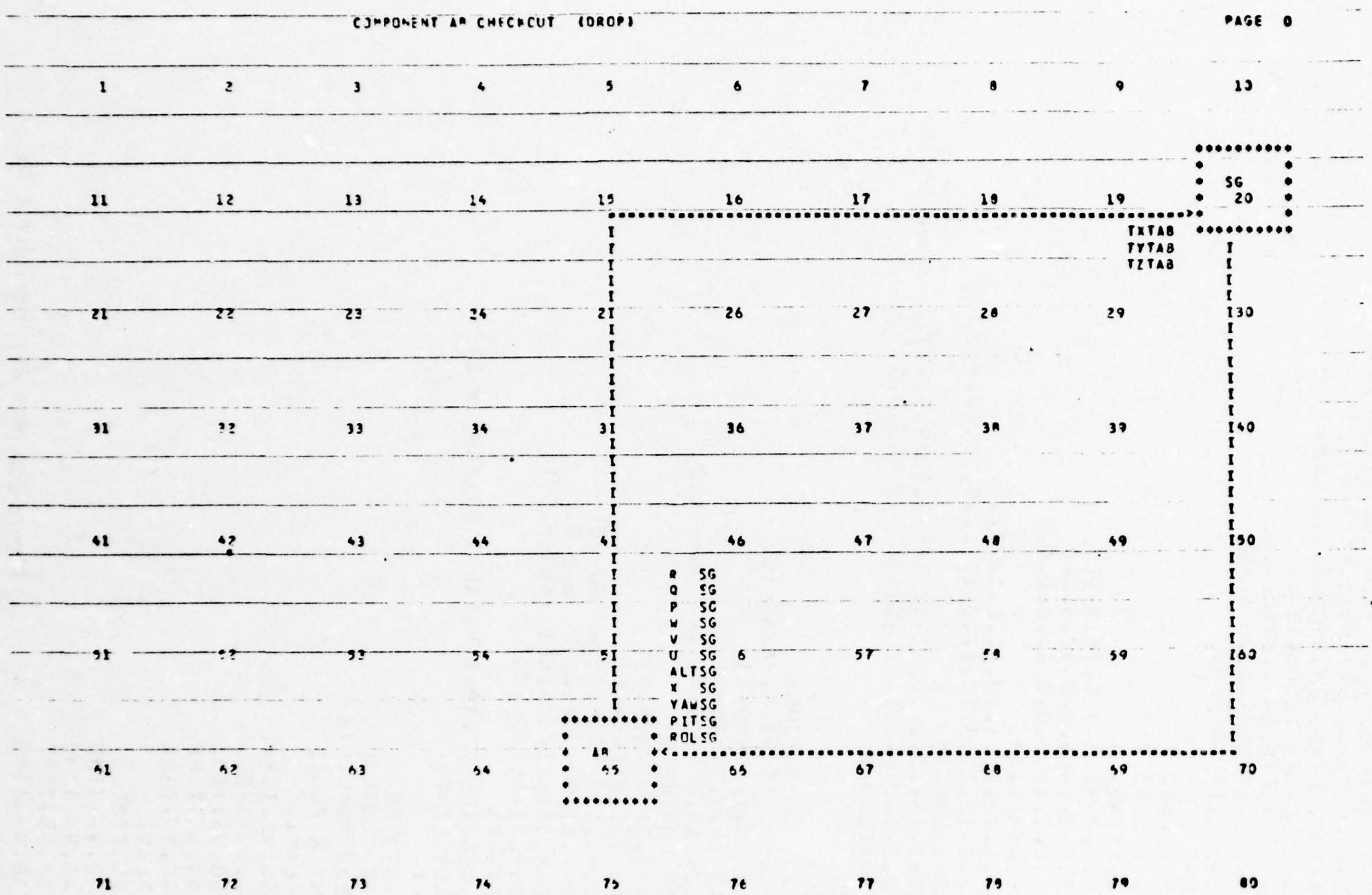


Figure 105 Air Bag Model Schematic

```
PARAMETER VALUES
AMASS=55.9,IXXSG=80,IYYSG=1240,IZZSG=900
NE AB=+9,NSTAB=1,NPTAB=10,PA AB=14.7
BSTAB=120,WLTAB=-15,BSCAB=63,WLCAB=0
CD1AB=.6,CD2AB=.2,CDAAB=.9
TAUAB=.005,EPCAB=1.,DMPAB=.02
WTRAB=25.,TTRAB=560,VU AB=6
WTLAB=25.,TTLAB=560.
AMOAB=0.
TABLE,ABLAB,3
10,0,36,0,90,0
TABLE,XYZAB,14
102,5.0,0,90,5.0,0,78,5.0,0
66.,5.0,0,54,5.0,0,42,5.0,0
30.,5.0,C,18,5.0,C,6.,5.C,0,0
TABLE,DSMAB,14
12,1,.1,12,1,.1,12,1,.1,12,1,.1
12,1,.5,12,1,.5,12,1,.5,12,1,.5,12,1,.5,0
TABLE,IALAB,18
1,.0061,13,10,1,.0061,13,10,1,.0061,13,10,1,.0061,13,10
1,0,0,0,1,0,0,0,1,C,0,0,1,C,0,0,1,0,C,0
TABLE,RELAB,2
2.,3.
0.,10.
INITIAL CONDITIONS
PITSG=5,ALTSG=3.4,YAWSG=10.
PTRAB=16.7,VTRAB=11
PTLAB=16.7,VTLAB=11
U  SG=168,W  SG=14.7,V  SG=-29.2
ERROR CONTROLS
PTRAB=.01,VTRAB=.01
PTLAB=.01,VTLAB=.01
U  SG=.01,V  SG=.01,W  SG=.01
P  SG=.01,Q  SG=.C1,R  SG=.01
ROLSG=.001,PITSG=.C01,YAWSG=.001
X  SG=.01,ALTSG=.0C1
NO STATES
INT CONTROL=PTRAB=1,VTRAB=1,PTLAB=1,VTLAB=1
STEADY STATE
XIC-X
ALL STATES
PRINT CONTROL=4
LINEAR ANALYSIS
PRINTER PLOTS
DISPLAY1
ALTSG,VS,TIME
R15,VS,TIME
PITSG,VS,TIME
ROLSG,VS,TIME
YAWSG,VS,TIME
DISPLAY2
R7,VS,TIME
FZTAB,VS,TIME
TYTAB,VS,TIME
FXTAB,VS,TIME
FYTAB,VS,TIME
DISPLAY3
TZTAB,VS,TIME
PTRAB,VS,TIME
PTLAB,VS,TIME
```

Figure 106 Air Bag Model Input Data

```
VTRAB,VS,TIME
WARAB,VS,TIME
DISPLAY4
R14,VS,TIME
TXTAB,VS,TIME
CPTAB,VS,TIME
TINC=.01,TMAX=1.2,PRATE=5,INT MODE=2
SIMULATE
```

Figure 106 Air-Bag Model Input Data (Concluded)

A tabular listing of the geometric information input into the AB component are shown in Table 7. The non-zero initial conditions are defined thus completing the model definition. Appropriate commands are made for the STEADY STATE and LINEAR ANALYSIS and then the vehicle drop with slideout is simulated. The results are shown in Table 8 and Figures 107 through 123.

A second simulation run successfully checked out the model in cross wind conditions.

A third simulation run successfully checked out the terrain model with (1-cosine) bump profile, AMODE=1.

A fourth simulation run successfully checked out the terrain model with random profile, AMODE=2.

TABLE 7 AIR BAG GEOMETRIC INPUT DATA ARRAY

***** AIR BAG PARAMETER DATA ***
*** FOR TWIN SYMMETRIC BAGS ***

ELEMENT	XA	YA	ZA	D	S	MU	IS	AP	LP	LM
1	102.00	10.00	0.00	12.00	1.000	.100	1	.00610	13.0	10.0
2	90.00	10.00	0.00	12.00	1.000	.100	1	.00610	13.0	10.0
3	78.00	10.00	0.00	12.00	1.000	.100	1	.00610	13.0	10.0
4	66.00	10.00	0.00	12.00	1.000	.100	1	.00610	13.0	10.0
5	54.00	10.00	0.00	12.00	1.000	.500	1	0.00000	0.0	0.0
6	42.00	10.00	0.00	12.00	1.000	.500	1	0.00000	0.0	0.0
7	30.00	10.00	0.00	12.00	1.000	.500	1	0.00000	0.0	0.0
8	18.00	10.00	0.00	12.00	1.000	.500	1	0.00000	0.0	0.0
9	6.00	10.00	0.00	12.00	1.000	.500	1	0.00000	0.0	0.0

TABLE 8 AIR BAG ELEMENT SECTION PROPERTIES

***** AIR BAG ELEMENT SECTION PROPERTIES **

*** DATA SET 1 *** MEMBRANE ELEMENT *****************

NPTS=10 DMU=.10 A= 10.00 B= 0.00 IC= 36.00 GA= 0.0 GB=90.0

FREE SHAPE ELEMENT PROPERTIES ZO= 13.03 YO= 5.00 LI= 18.00 AS= 162.3

*** ZO ARRAY ***

13.03	11.72	10.42	9.12	7.82	6.51	5.21	3.91	2.61	1.30

THE FOLLOWING ARRAYS CORRESPOND COLUMN-TO-ELEMENT WITH THE ZO ARRAY
ROWS CORRESPOND TO INCREASING MAGNITUDES OF MU BEGINNING AT MU=0
DATA GENERATED FOR POSITIVE MU STORED IN 1ST ARRAY
DATA GENERATED FOR NEGATIVE MU STORED IN 2ND ARRAY

*** YO ARRAY ***

5.00	3.38	1.95	.70	-.74	-2.76	-4.82	-6.86	-8.91	-10.95
5.00	2.95	.88	-1.41	-3.38	-5.25	-7.03	-8.71	-10.29	-11.74
5.00	2.41	.63	-2.01	-3.92	-5.71	-7.39	-8.97	-10.44	-11.80
5.00	2.07	-.33	-2.35	-4.22	-5.95	-7.58	-9.10	-10.52	-11.83
5.00	1.[illegible]	-.58	-2.58	-4.41	-6.11	-7.70	-9.18	-10.56	-11.85
5.00	1.59	-.77	-2.74	-4.55	-6.22	-7.78	-9.24	-10.60	-11.86
5.00	1.47	-.91	-2.87	-4.66	-6.31	-7.85	-9.28	-10.62	-11.87
5.00	1.28	-1.02	-2.97	-4.74	-6.38	-7.90	-9.31	-10.64	-11.87
5.00	1.17	-1.11	-3.05	-4.81	-6.43	-7.94	-9.34	-10.66	-11.86
5.00	1.07	-1.19	-3.12	-4.87	-6.48	-7.97	-9.37	-10.67	-11.88

5.00	3.38	1.95	.70	-.74	-2.76	-4.82	-6.86	-8.91	-10.95
5.00	3.95	3.54	3.92	4.54	4.97	5.12	4.96	4.44	3.40
5.00	4.50	4.69	5.21	5.66	5.89	5.80	5.58	4.93	3.76
5.00	4.99	5.50	6.06	6.43	6.57	6.46	6.07	5.34	4.07
5.00	5.42	6.11	6.63	7.02	7.10	6.94	6.49	5.70	4.35
5.00	5.76	6.60	7.19	7.49	7.55	7.34	6.84	6.01	4.59
5.00	6.19	7.00	7.59	7.89	7.92	7.69	7.16	6.28	4.82
5.00	6.36	7.33	7.93	8.23	8.25	8.00	7.44	6.54	5.02
5.00	6.59	7.61	8.23	8.52	8.53	8.27	7.69	6.76	5.22
5.00	6.79	7.86	8.49	8.78	8.79	8.51	7.92	6.98	5.39

*** LI ARRAY ***

18.00	16.38	14.55	13.70	12.26	10.22	8.18	6.14	4.09	2.05
18.00	16.75	14.45	12.59	10.81	9.11	7.48	5.90	4.33	2.73
18.00	16.20	14.39	12.69	11.10	9.58	8.10	6.64	5.15	3.47
18.00	16.18	14.46	12.90	11.44	10.05	8.66	7.29	5.81	4.05
18.00	16.19	14.57	13.12	11.77	10.47	9.18	7.84	6.37	4.54
18.00	16.22	14.69	13.34	12.07	10.85	9.62	8.32	6.85	4.96
18.00	16.25	14.81	13.53	12.35	11.19	10.01	8.75	7.28	5.34
18.00	16.29	14.92	13.72	12.60	11.50	10.37	9.13	7.67	5.67
18.00	16.33	15.03	13.89	12.62	11.76	10.69	9.48	8.02	5.98
18.00	16.35	15.13	14.04	13.03	12.04	10.99	9.80	8.35	6.27

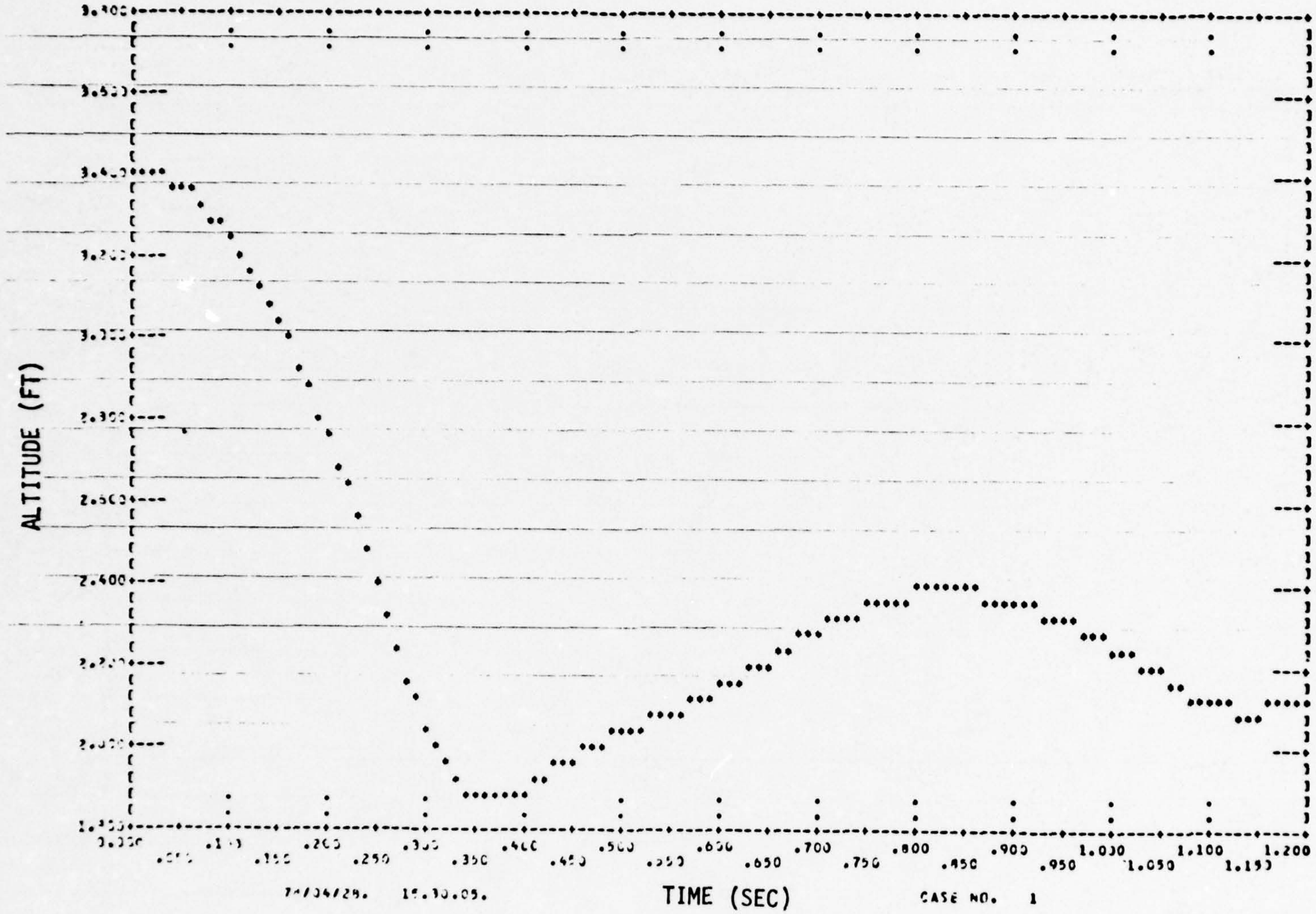


Figure 107 Time History Plot of Air Bag Simulation, Altitude

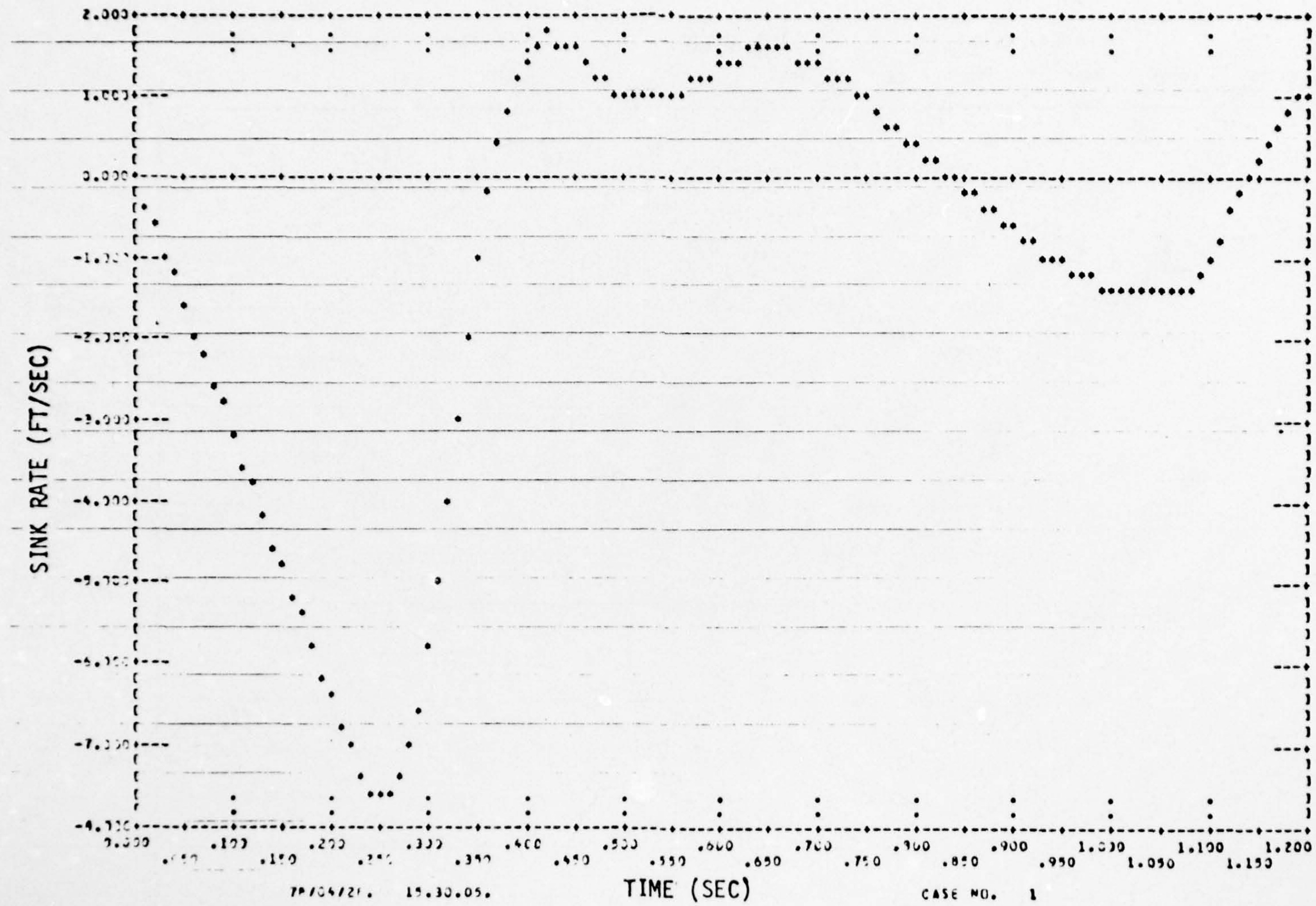


Figure 108 Time History Plot of Air Bag Simulation, Sink Rate

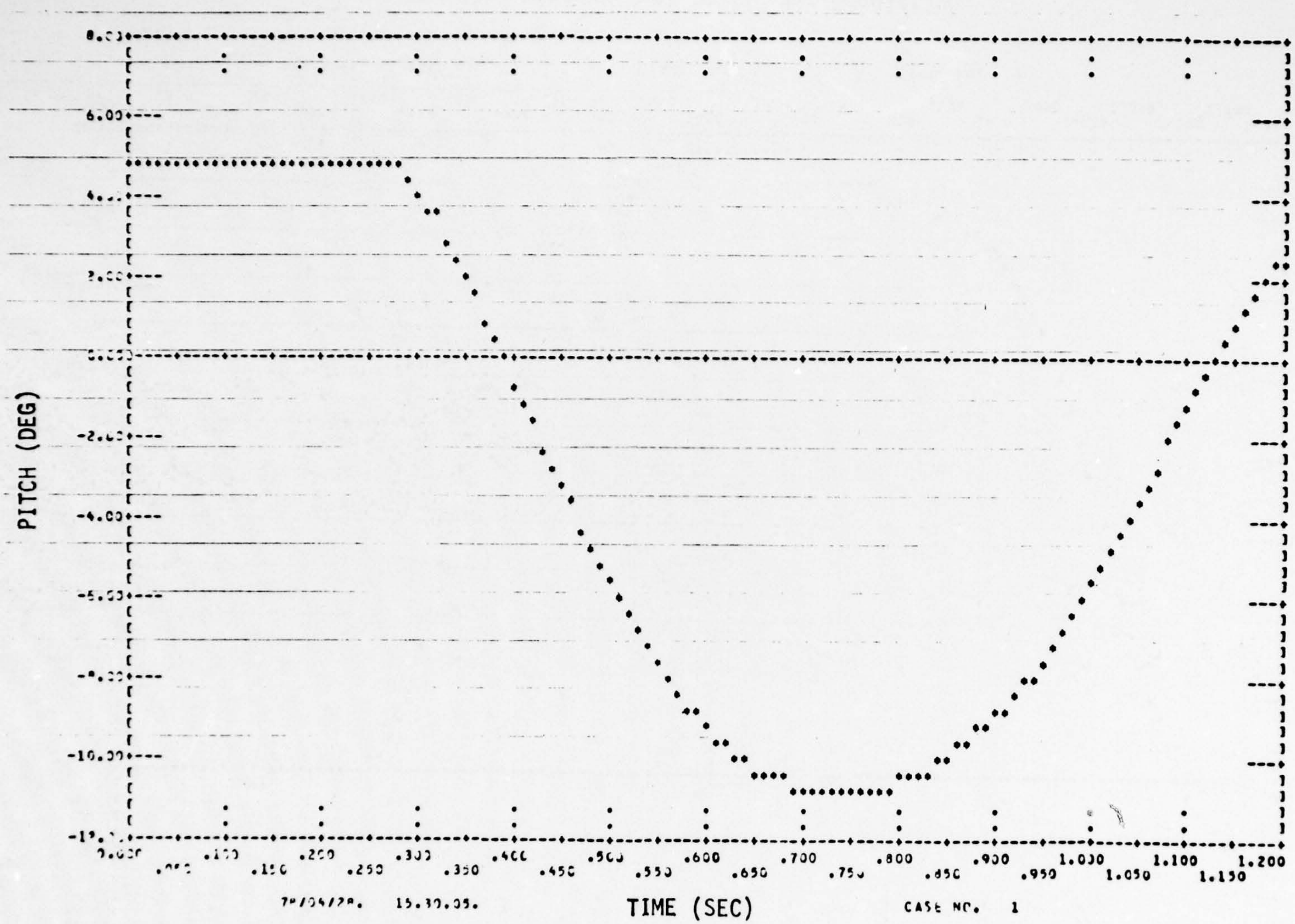


Figure 109 Time History Plot of Air Bag Simulation,
Pitch Angle

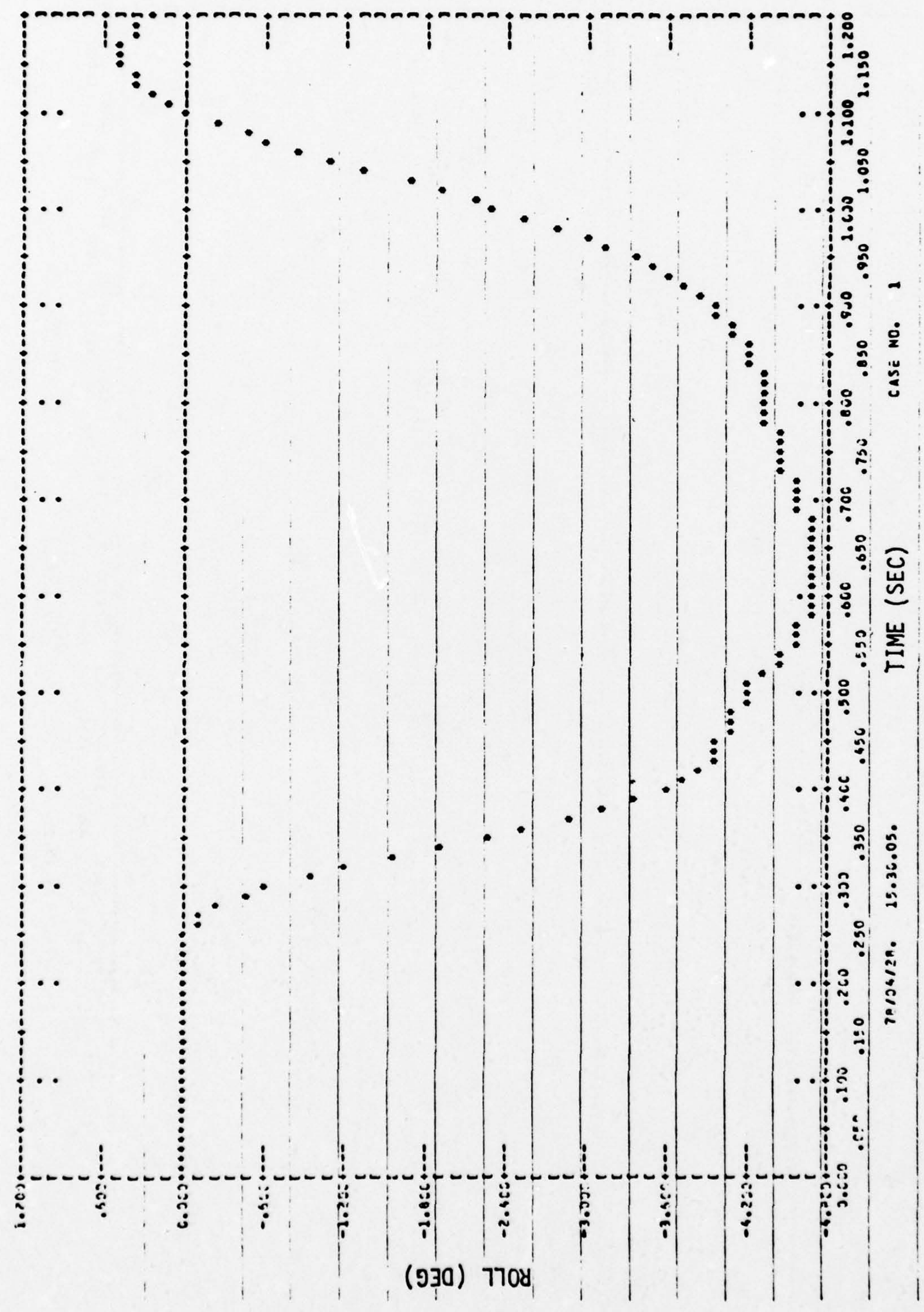


Figure 110 Time History Plot of Air Bag Simulation, Roll Angle

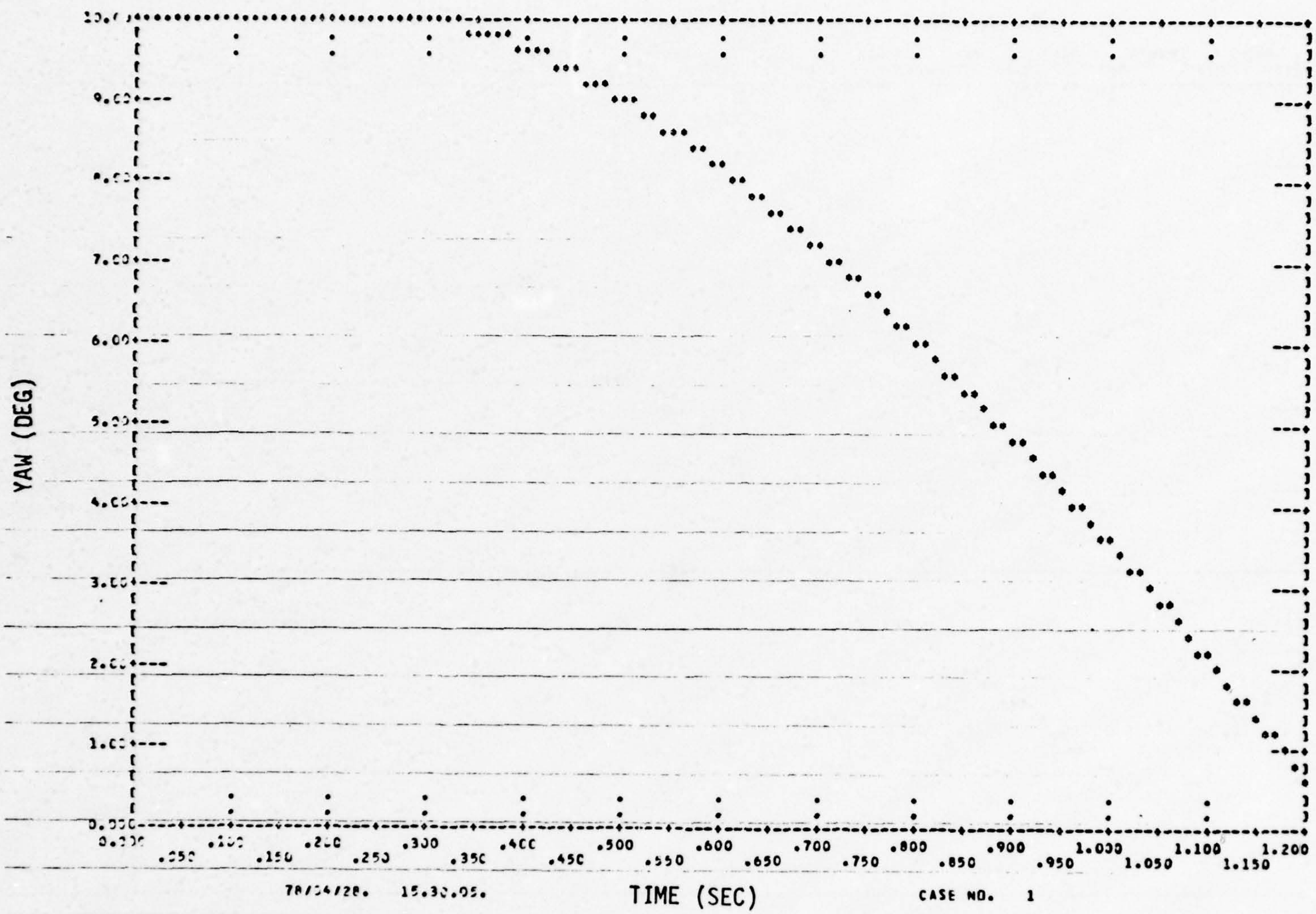


Figure 111 Time History Plot of Air Bag Simulation, Yaw Angle

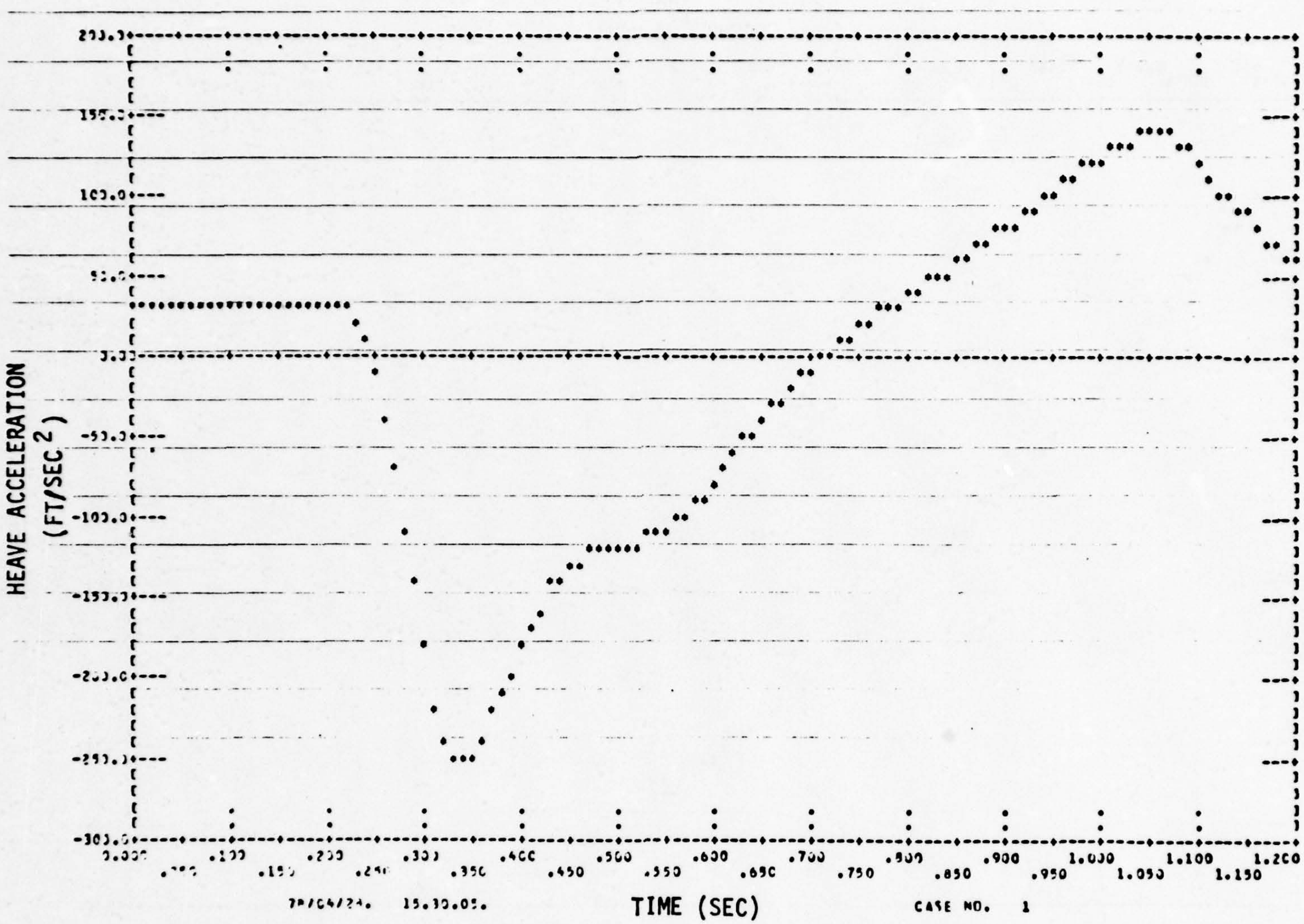


Figure 112 Time History Plot of Air Bag Simulation, Heave Accel.

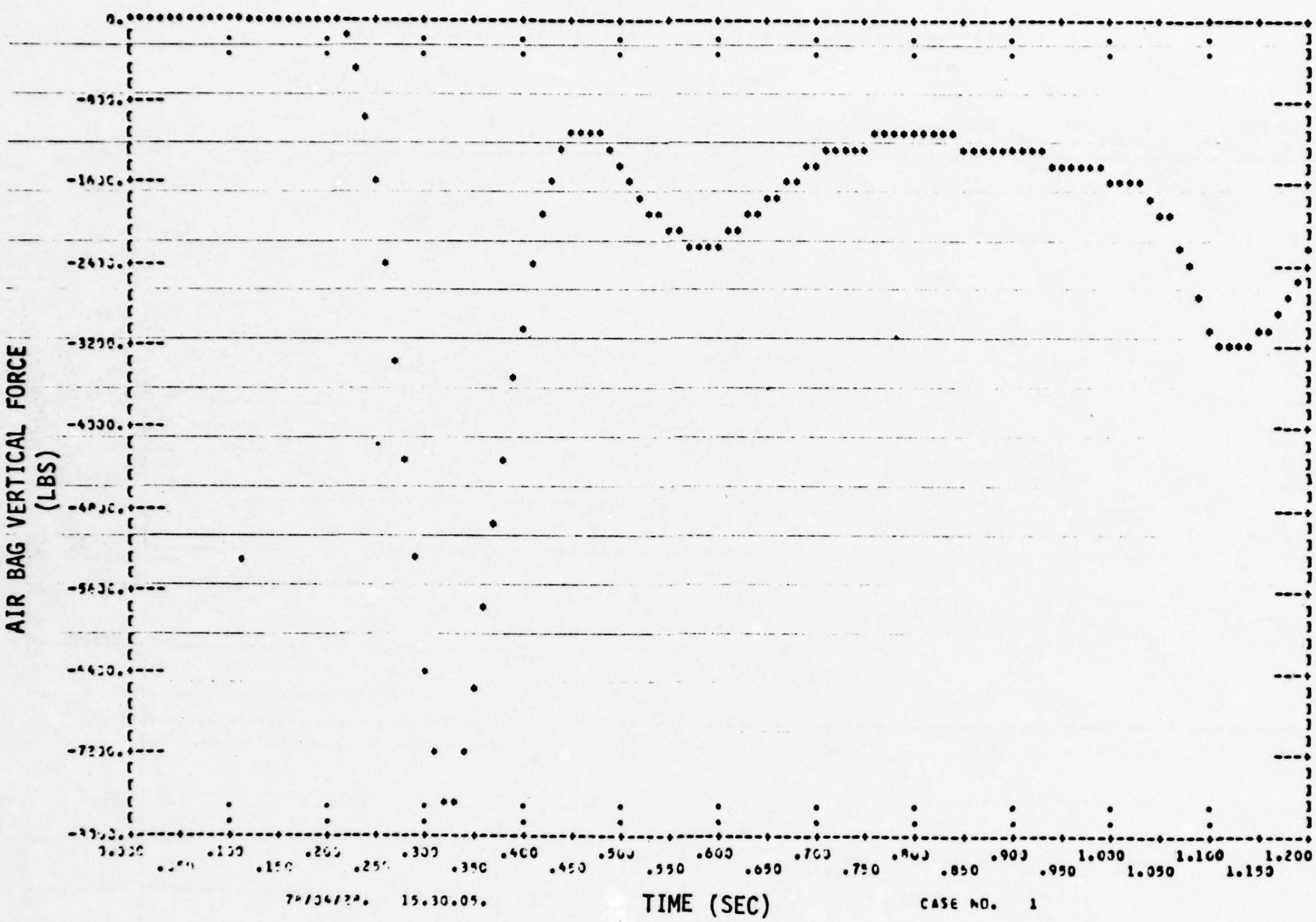


Figure 113 Time History Plot of Air Bag Simulation, Bag Vertical Force

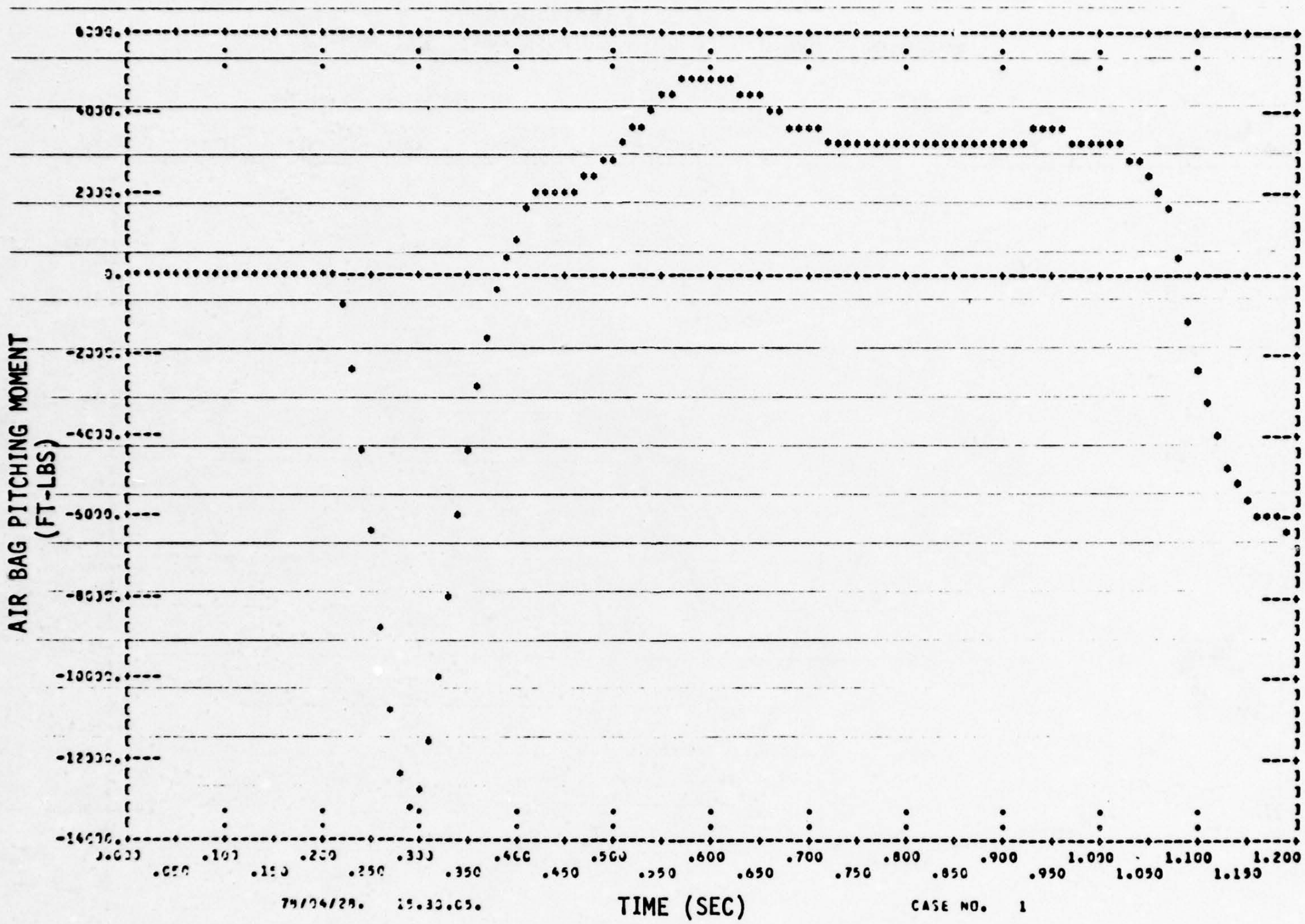


Figure 114 Time History Plot of Air Bag Simulation,
Bag Pitch Moment

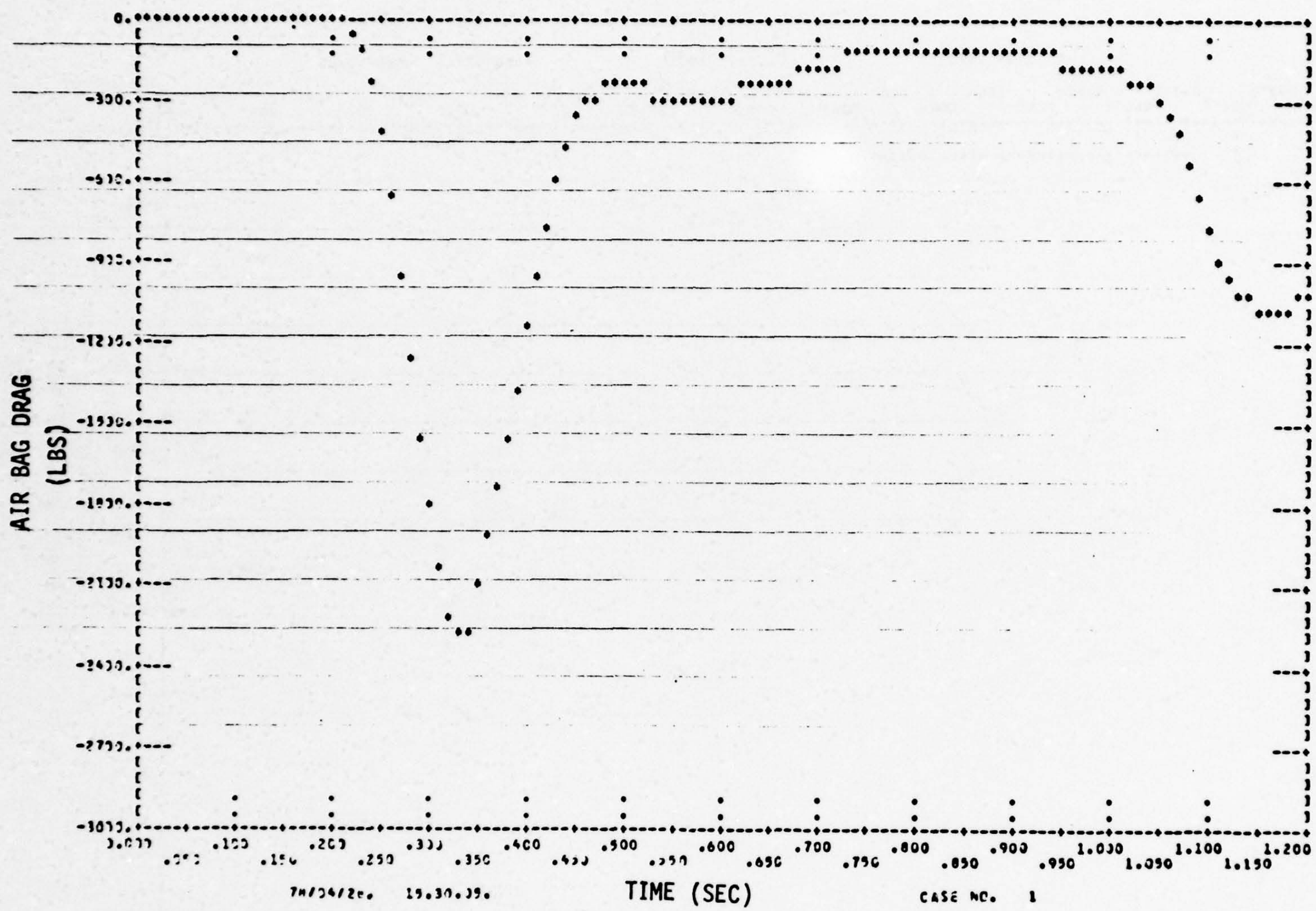


Figure 115 Time History Plot of Air Bag Simulation,
Bag Drag

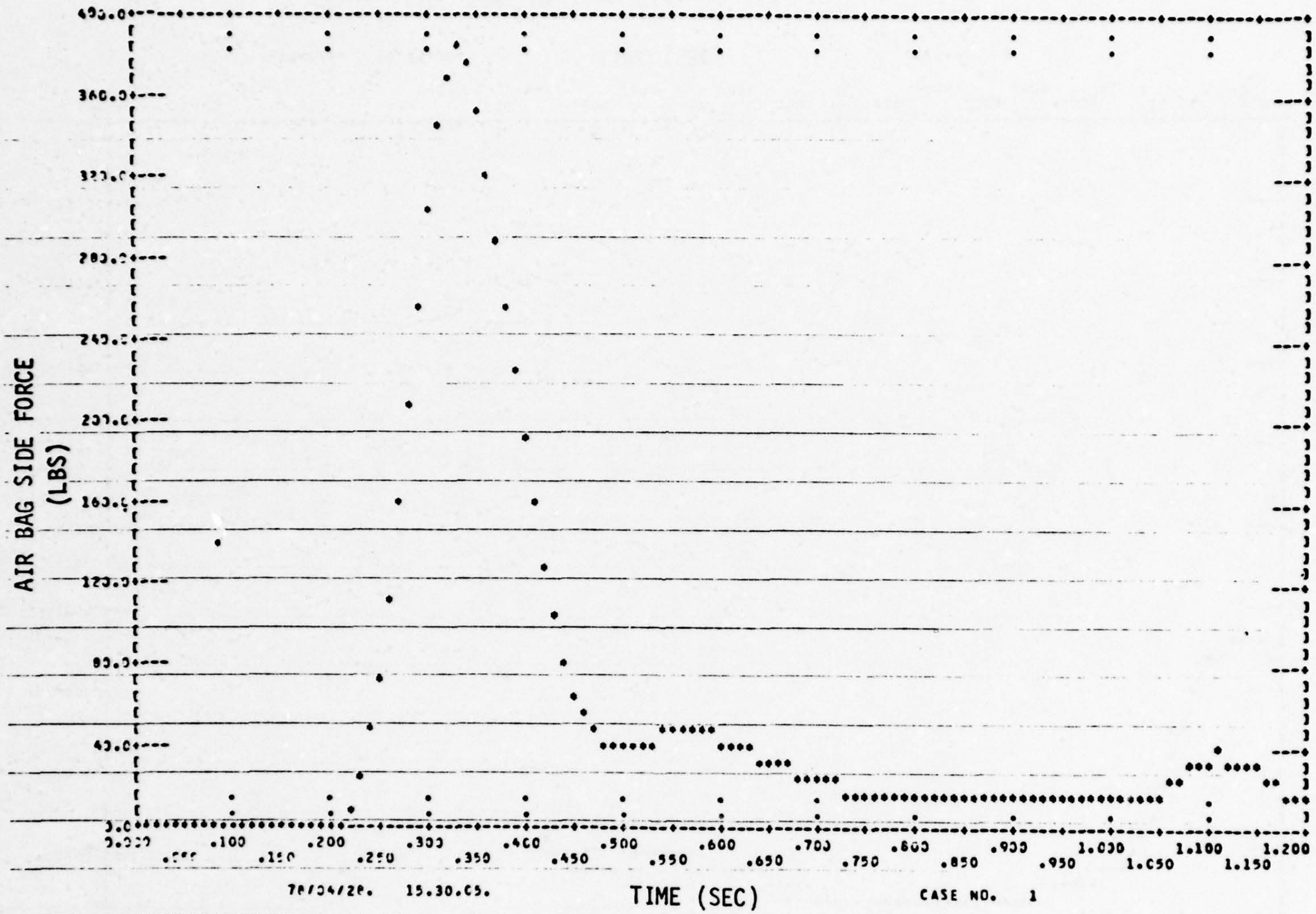


Figure 116 Time History Plot of Air Bag Simulation, Bag Side Force

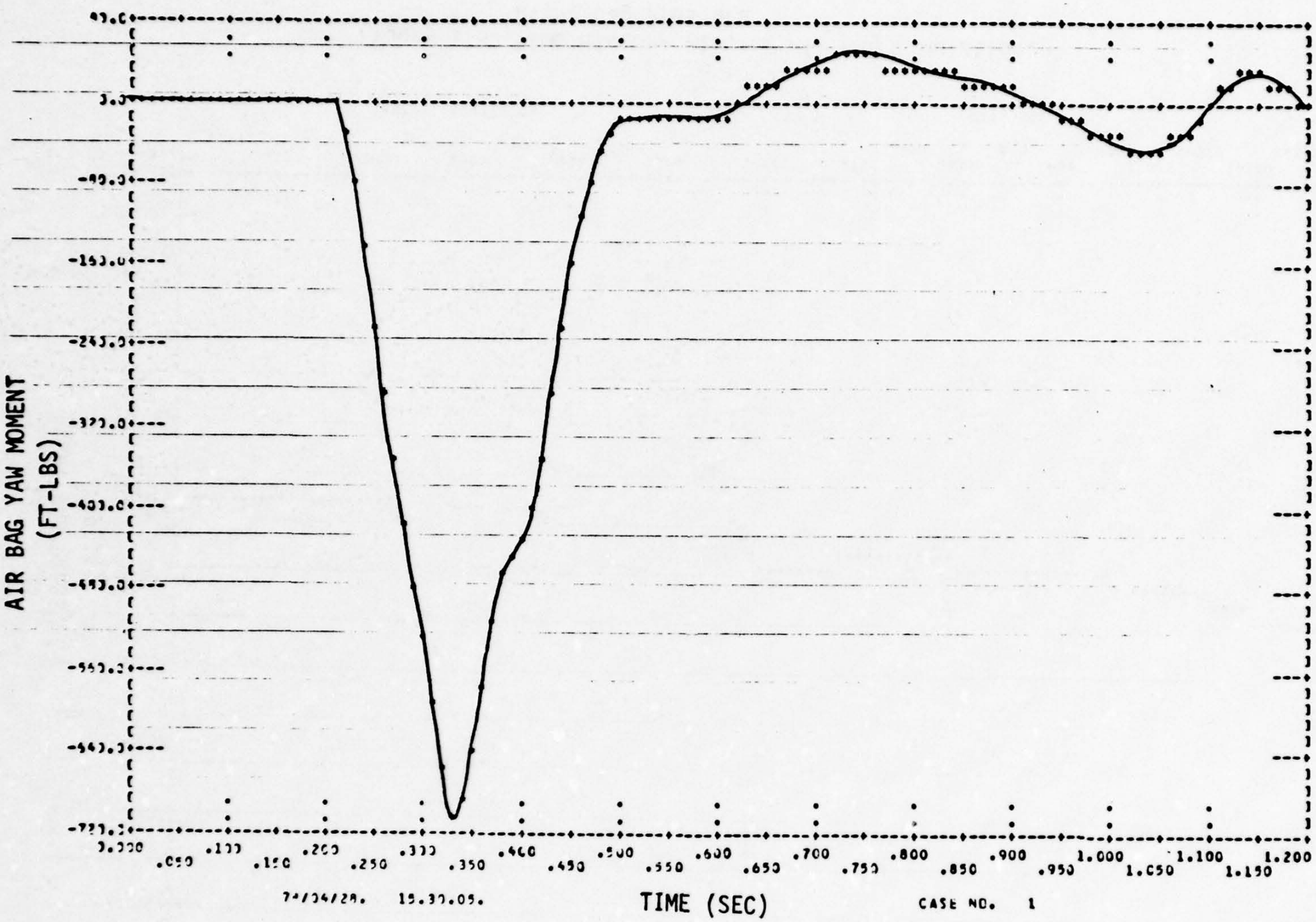


Figure 117 Time History Plot of Air Bag Simulation, Bag Yaw Moment

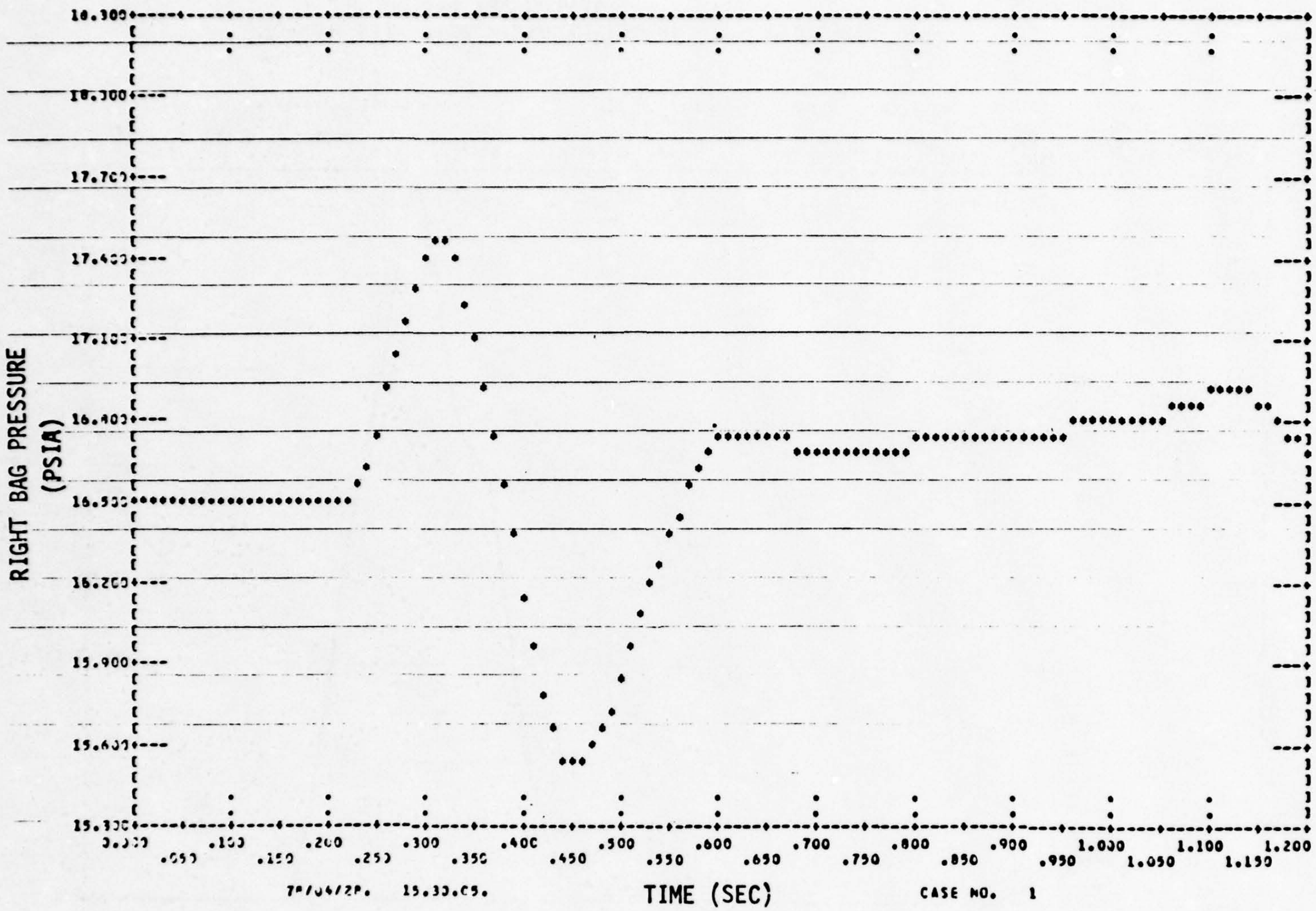


Figure 118 Time History Plot of Air Bag Simulation, Right Bag Pressure

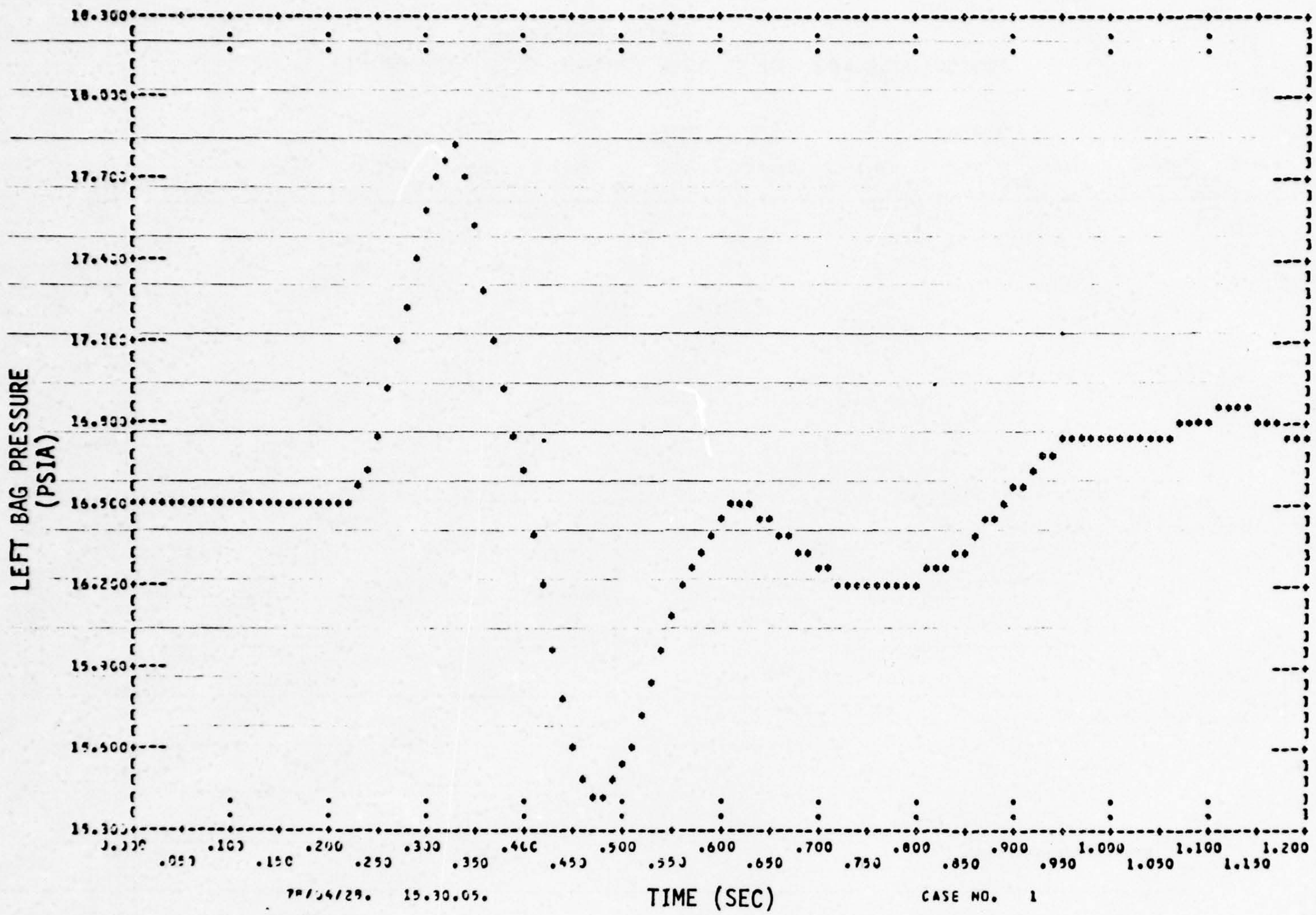


Figure 119 Time History Plot of Air Bag Simulation, Left Bag Pressure

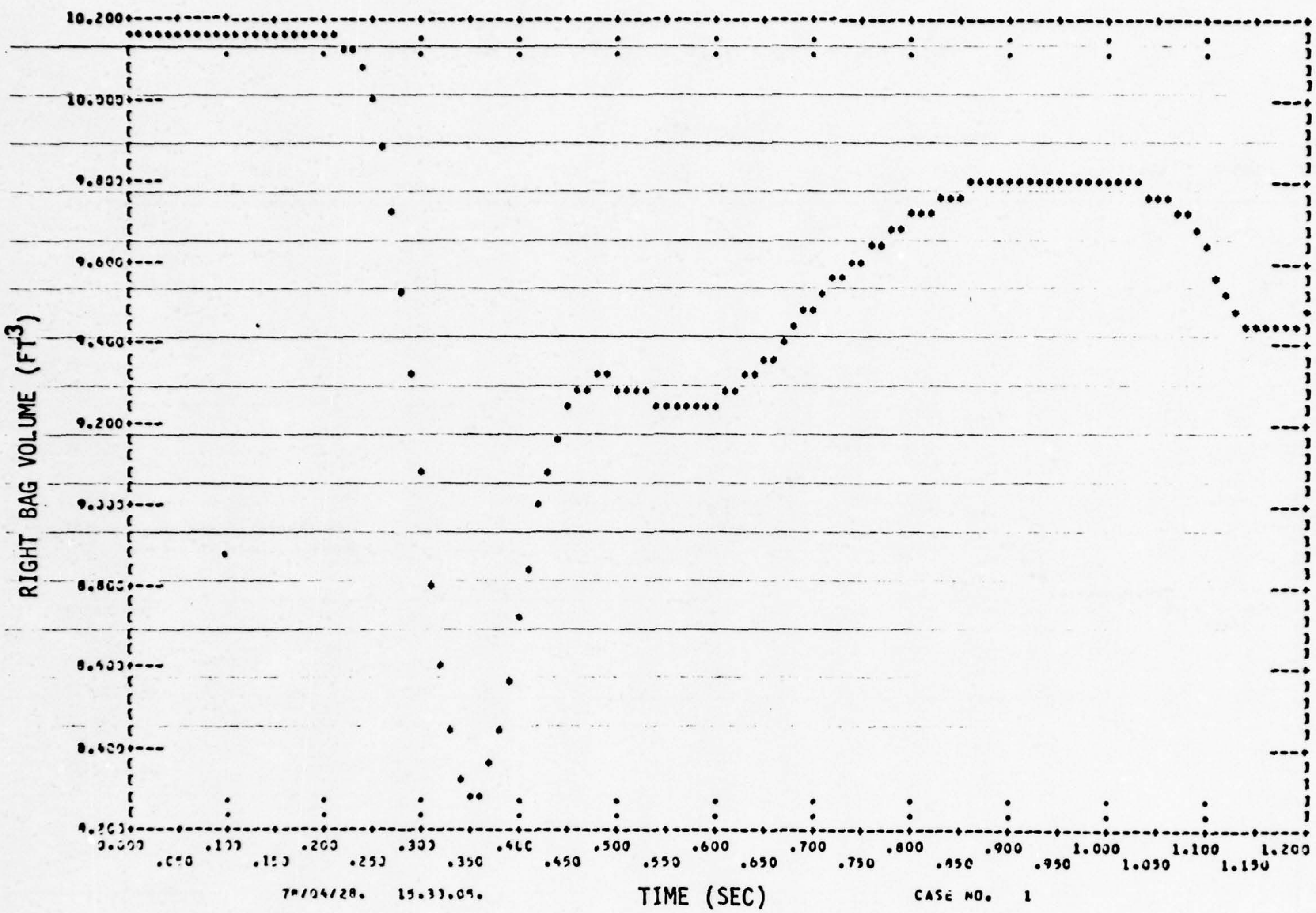


Figure 120 Time History Plot of Air Bag Simulation,
Right Bag Volume

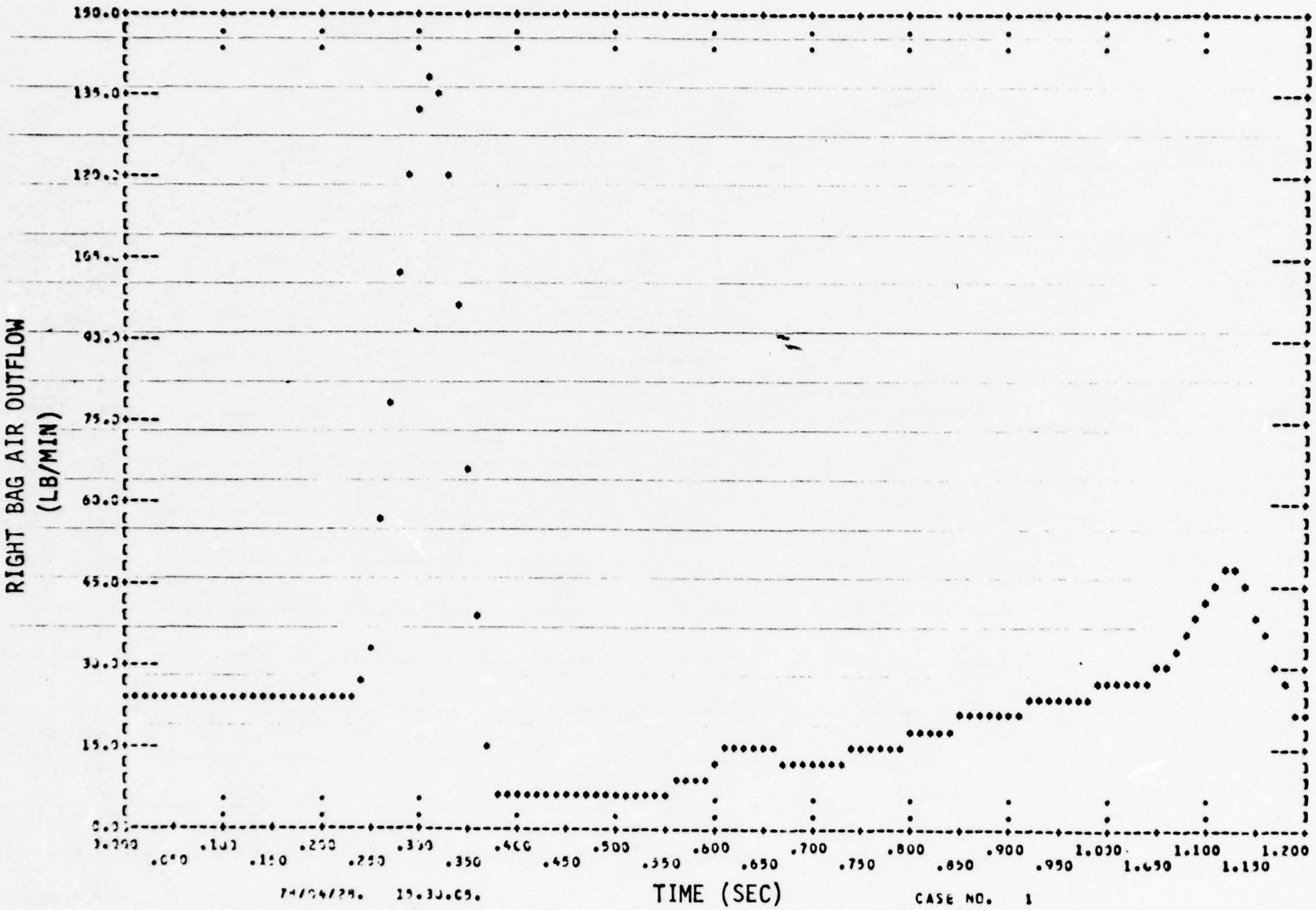


Figure 121 Time History Plot of Air Bag Simulation, Right Bag Air Outflow

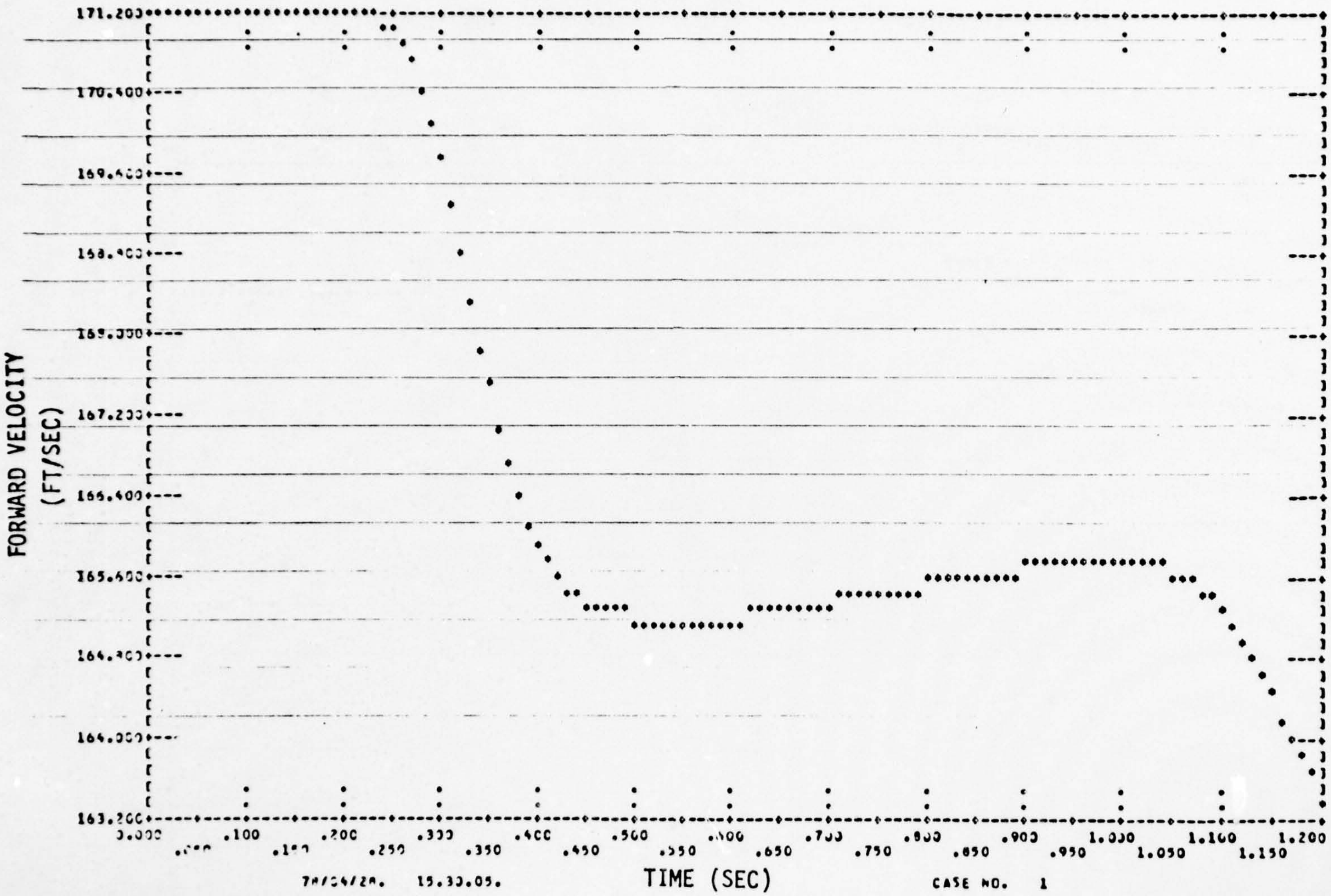


Figure 122 Time History Plot of Air Bag Simulation, Forward Velocity

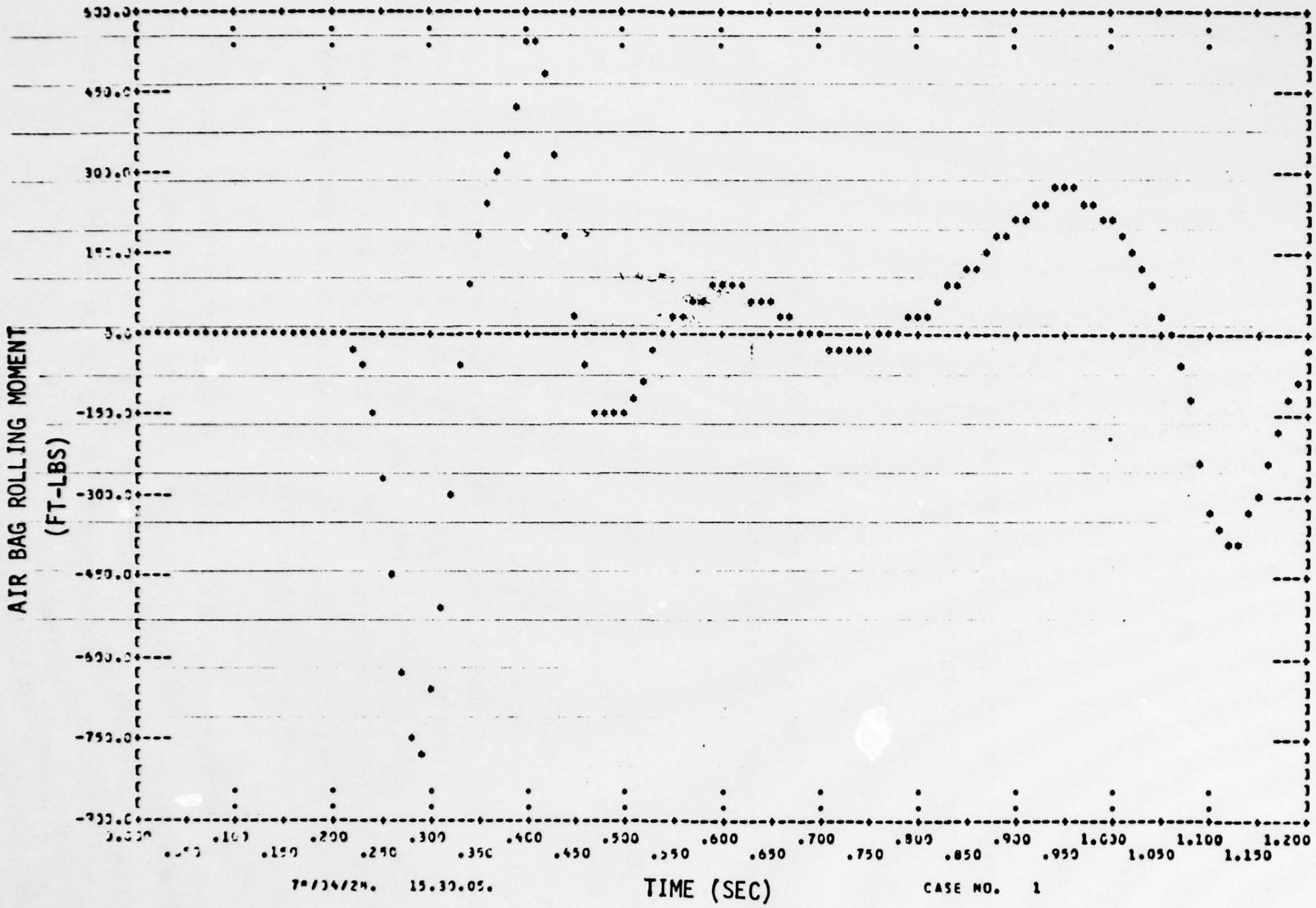


Figure 123 Time History Plot of Air Bag Simulation, Bag Roll Moment

SECTION VII

ARRESTING GEAR SIMULATION

7.1 Objectives

The objective of the arresting gear simulation was to checkout and debug the arresting system model component "AS".

7.2 Technical Approach

Under Task 3, an arresting gear component was developed to simulate arresting hook and cable dynamics of a water twister energy absorber type of system, Figure 124.

The Water Twister (Registered Trademark) is a simple water brake that converts kinetic energy to heat through turbulence. The brake consists of a fluid filled steel casing, with internal stator vanes, which houses a vaned centrifugal rotor. The rotor is mounted on a shaft which extends out of the top of the casing. A storage reel for a nylon tape purchase element is mounted on and splined to the top end of the rotor shaft. The tape is wrapped on this storage reel, layer on layer, forming a spiral wrap. Pulling the tape off the reel causes the shaft and vaned rotor to revolve within the fluid filled casing, creating turbulence. For the derivation of model equations and other details, see Volume I, Section VIII.

The component "AS" and associated subroutines and functions were successfully used along with the 6 DOF components "SG" in a cable-hook-engagement, impact and slideout simulation. The vehicle (spring-mass) configuration for this checkout simulation is shown in Figure 125. System parametric data selected for the analysis are as follows:

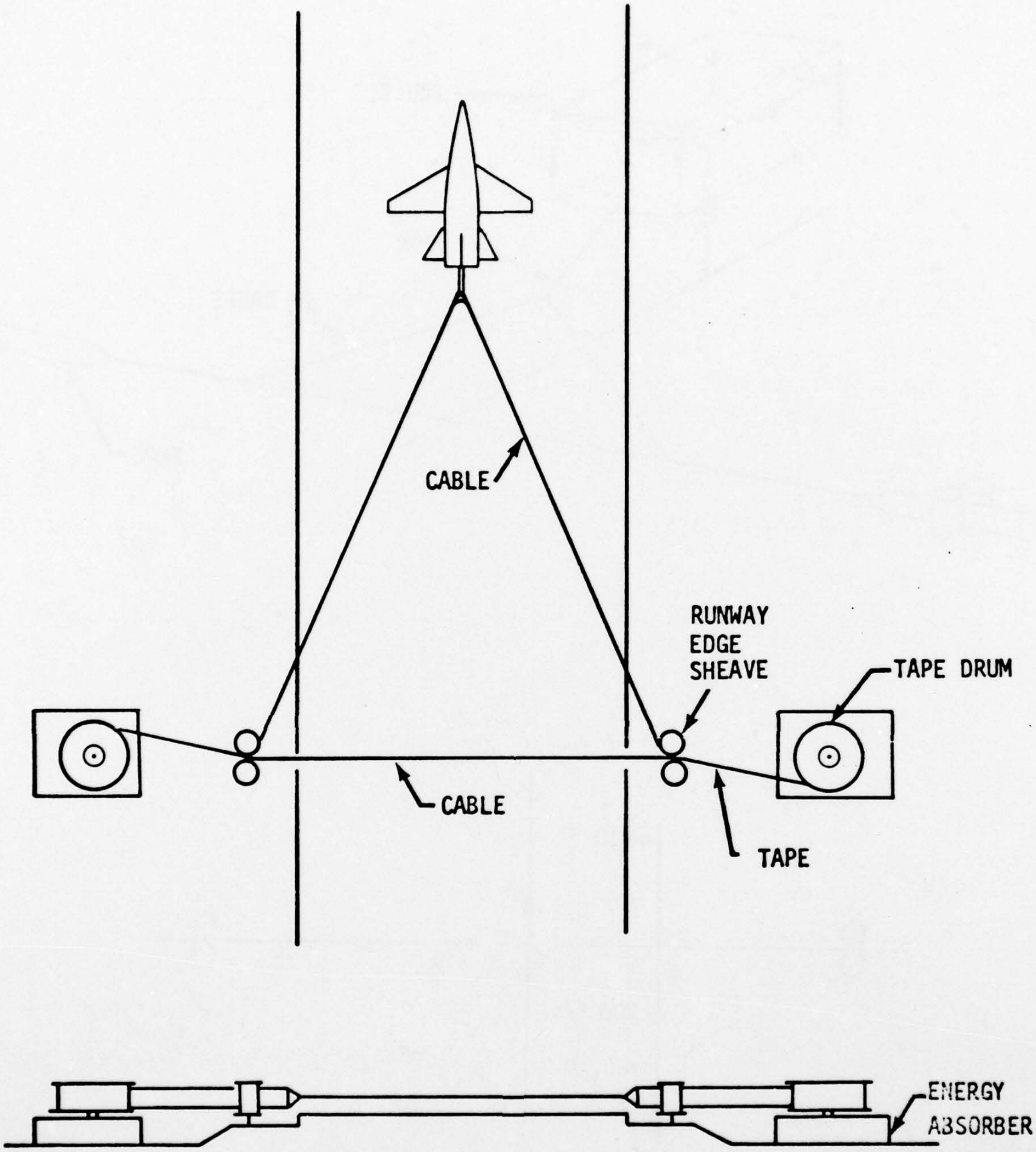
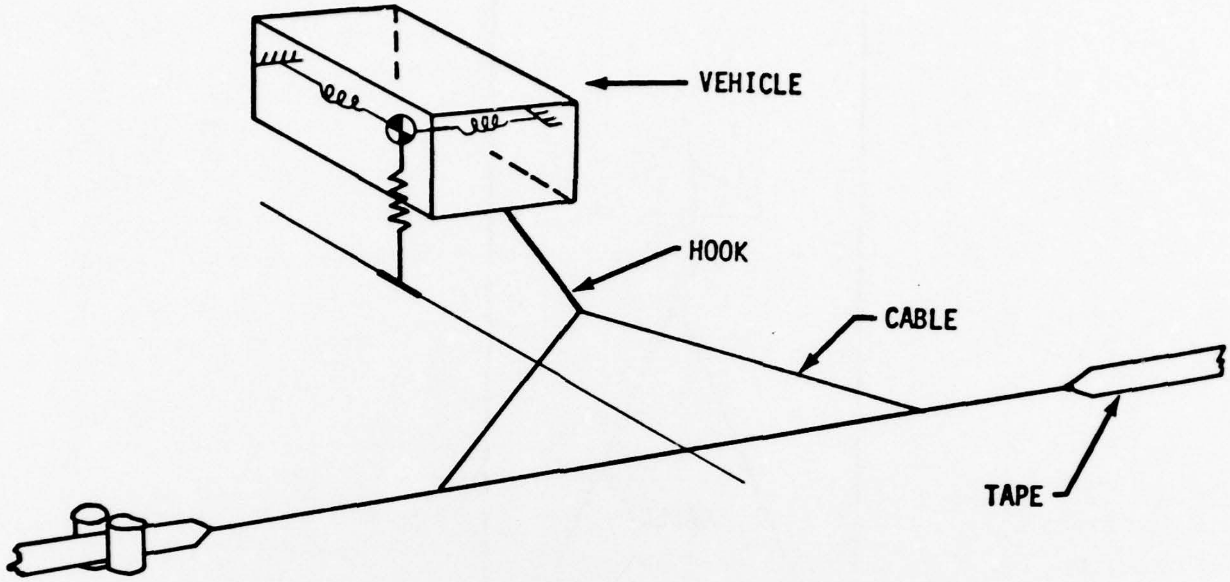


Figure 124 Water Twister Arresting System



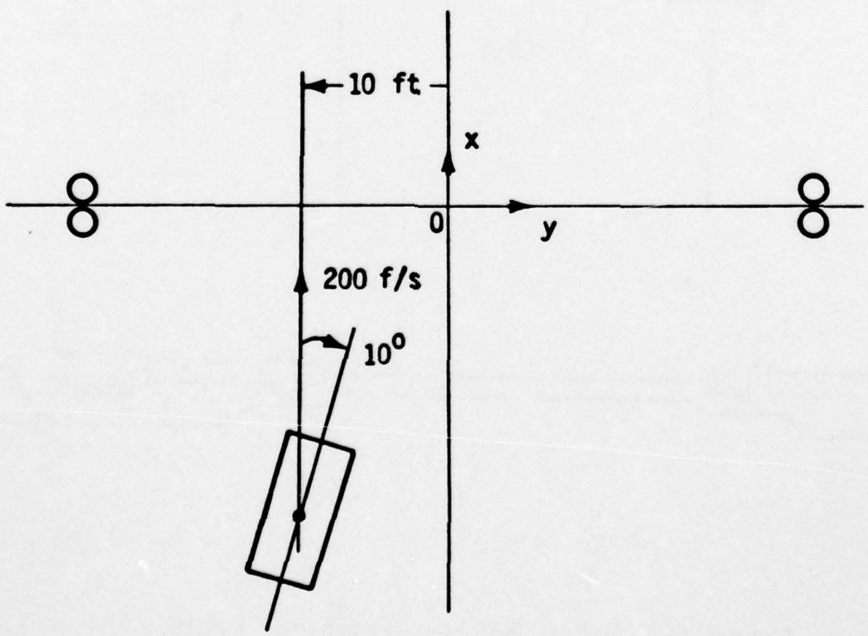


Figure 125 Arresting System Simulation Configuration

Vehicle weight = 2900 lbs.

I_{xx} = 1200 slug ft^2

I_{yy} = 1800 slug ft^2

I_{zz} = 2800 slug ft^2

Cable pickup velocity = 200 fps

Hook length = 20 in.

Runway span = 100 ft.

Cable height = 6 in.

Yaw attitude = 10 deg

7.3 Simulation Description and Results

The basic modules required to model the vehicle and arresting system dynamics are components SG and AS (no aerodynamic effects are simulated). A description of the model is shown in Figure 126. The EASY generated model schematic is shown in Figure 127.

The input data used to simulate this model are shown in Figure 128. Vehicle mass and spring rate data and the moments of inertias are input followed by vehicle c.g. and hook pivot locations and various geometric data for cable. The table ET AS defines the tape stress as a function of strain. The non-zero INITIAL CONDITIONS are defined and appropriate commands for time-history plots are made. Finally the TINC and SIMULATE commands generate the time history simulations and the results are shown in Figures 129 through 134.

The total hook load (THL) is shown in the top chart of Figure 131. The next three time histories are the component loads contributing to the THL: the cable impact load (CIL) caused by the initial pickup of the cable mass, and the right and left cable/tape tension (TR and TL respectively).

The step in the CIL is caused by the off-center engagement of the hook and cable. The model computes the change in momentum due to cable mass pickup until the kink wave reaches the runway edge sheaves which in this case occurs for the left portion of the cable before the right portion.

```
MODEL DESCRIPTION         COMPONENT AS CHECKOUT
ADD PARAMETERS=AMASS,AKZ,AKR,AKP
LOCATION=05     AS     INPUTS=SG
FORTRAN STATEMENTS
      FZ2S2=AKZ*(ALTSG-.08333*WLCAS)-1700.*W   SG-32.2*AMASS
      TX2S2=-.01745*AKR*ROLSG-385.*P   SG
      TY2S2=-.01745*AKP*PITSG-385.*Q   SG
LOCATION=15     S2
INPUTS=AS(FX=FX,1,FY=FY,1,FZ=FZ,1,TX=TX,1,TY=TY,1,TZ=TZ,1)
FORTRAN STATEMENTS
      UD SG=FX3S2/AMASS-(Q   SG*W   SG-R   SG*V   SG)*.01745-
     1                        32.2*SIN(PITSG*.01745)
      VD SG=FY3S2/AMASS-(R   SG*U   SG-P   SG*W   SG)*.01745
     1            +32.2*COS(PITSG*.01745)*SIN(ROLSG*.01745)
      WD SG=FZ3S2/AMASS-(P   SG*V   SG-Q   SG*U   SG)*.01745
     1            +32.2*COS(PITSG*.01745)*COS(ROLSG*.01745)
LOCATION=20     SG     INPUTS=S2(TX,3=TX,TY,3=TY,TZ,3=TZ)
END OF MODEL
PRINT
```

Figure 126 Arresting System Model Description

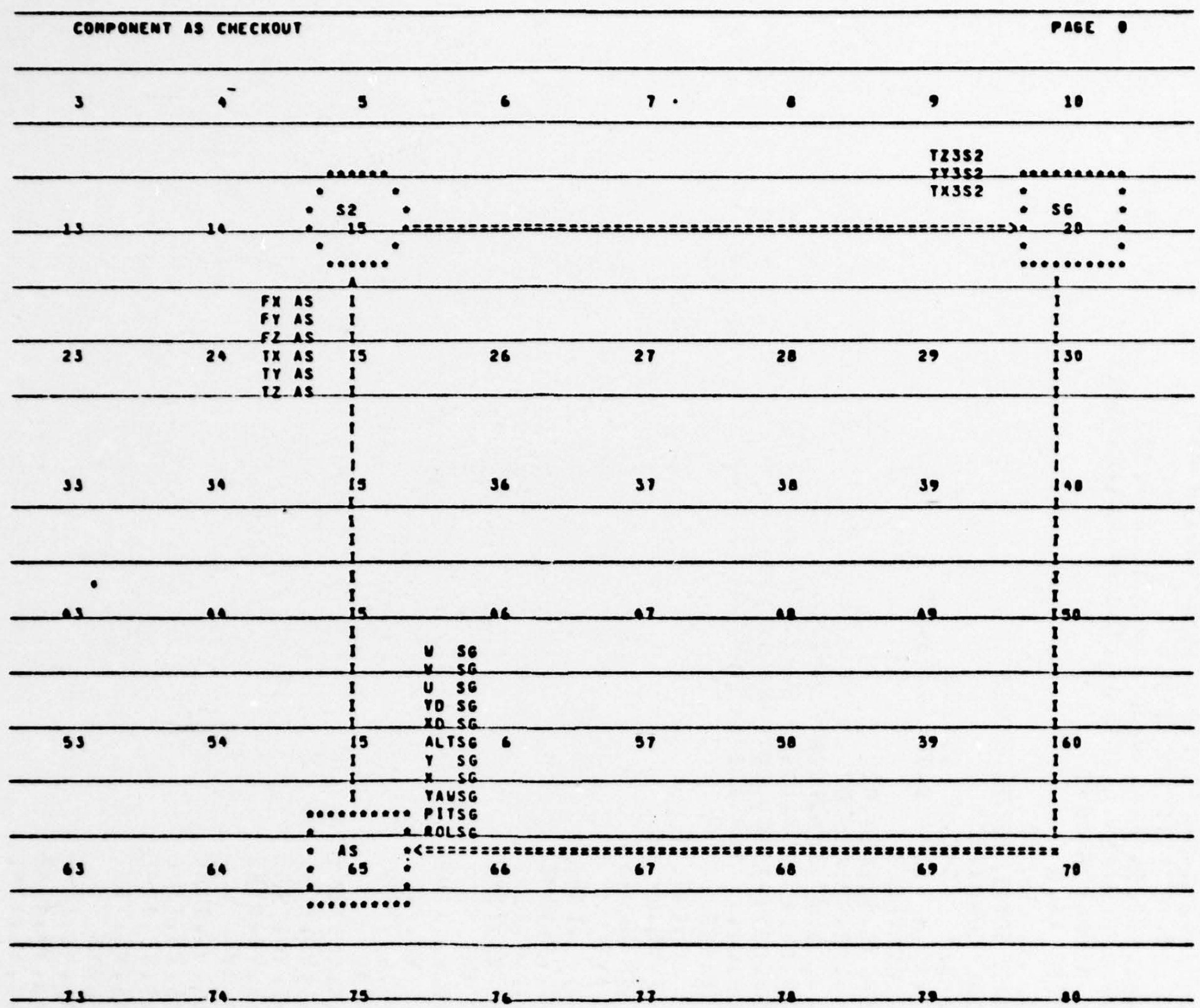


Figure 127 Arresting System Model Schematic

```
PARAMETER VALUES
AMASS=90,AKZ=320C0,AKR=4C0000,AKP=280000,VO AS=200
IXXSG=1200,IYYSG=1800,IZZSG=2800
FX2S2=0,FY2S2=0,TZ2S2=0
BSCAS=0,WLCAS=36,BSHAS=80,WLHAS=20,LH AS=20
YS AS=100,YM AS=10,HC AS=.5
EC AS=1.2E7,DNCAS=.25,AC AS=.2,ICSAS=2500
DNTAS=.03,THKAS=.1,WDTAS=5,TPCAS=400,RO AS=13
IDRAS=30000,DMPAS=3
TABLE,ET AS,5
0, .05, .1, .15, .2
0, 22446, 50443, 85272, 128210
INITIAL CONDITIONS
ALTSG=3,X  SG=C,U  SG=200
YAWSG=10,Y  SG=-10,V  SG=-34.7
ERROR CONTROLS
G1RAS=.01,G1LAS=.01
G2RAS=.01,G2LAS=.01
U  SG=.01,V  SG=.C1,W  SG=.01
P  SG=.01,R  SG=.C1,Q  SG=.01
ROLSG=.001,PITSG=.001,YAWSG=.001
X  SG=.01,Y  SG=.CC1,ALTSG=.001
ALL STATES
PRINT CONTROL=4
LINEAR ANALYSIS
PRINTER PLOTS
DISPLAY1
X  SG,VS,TIME
Y  SG,VS,TIME
ALTSG,VS,TIME
XD SG,VS,TIME
DISPLAY2
ROLSG,VS,TIME
PITSG,VS,TIME
YAWSG,VS,TIME
DISPLAY3
THLAS,VS,TIME
CILAS,VS,TIME
TR AS,VS,TIME
TL AS,VS,TIME
DISPLAY4
G1RAS,VS,TIME
G2RAS,VS,TIME
G1LAS,VS,TIME
G2LAS,VS,TIME
DISPLAY5
FX AS,VS,TIME
FY AS,VS,TIME
FZ AS,VS,TIME
DISPLAY6
TX AS,VS,TIME
TY AS,VS,TIME
TZ AS,VS,TIME
TINC=.01,TMAX=1.2,PRATE=5,INT MODE=2
SIMULATE
XIC-X
LINEAR ANALYSIS
```

Figure 128 Arresting System Model Data Input

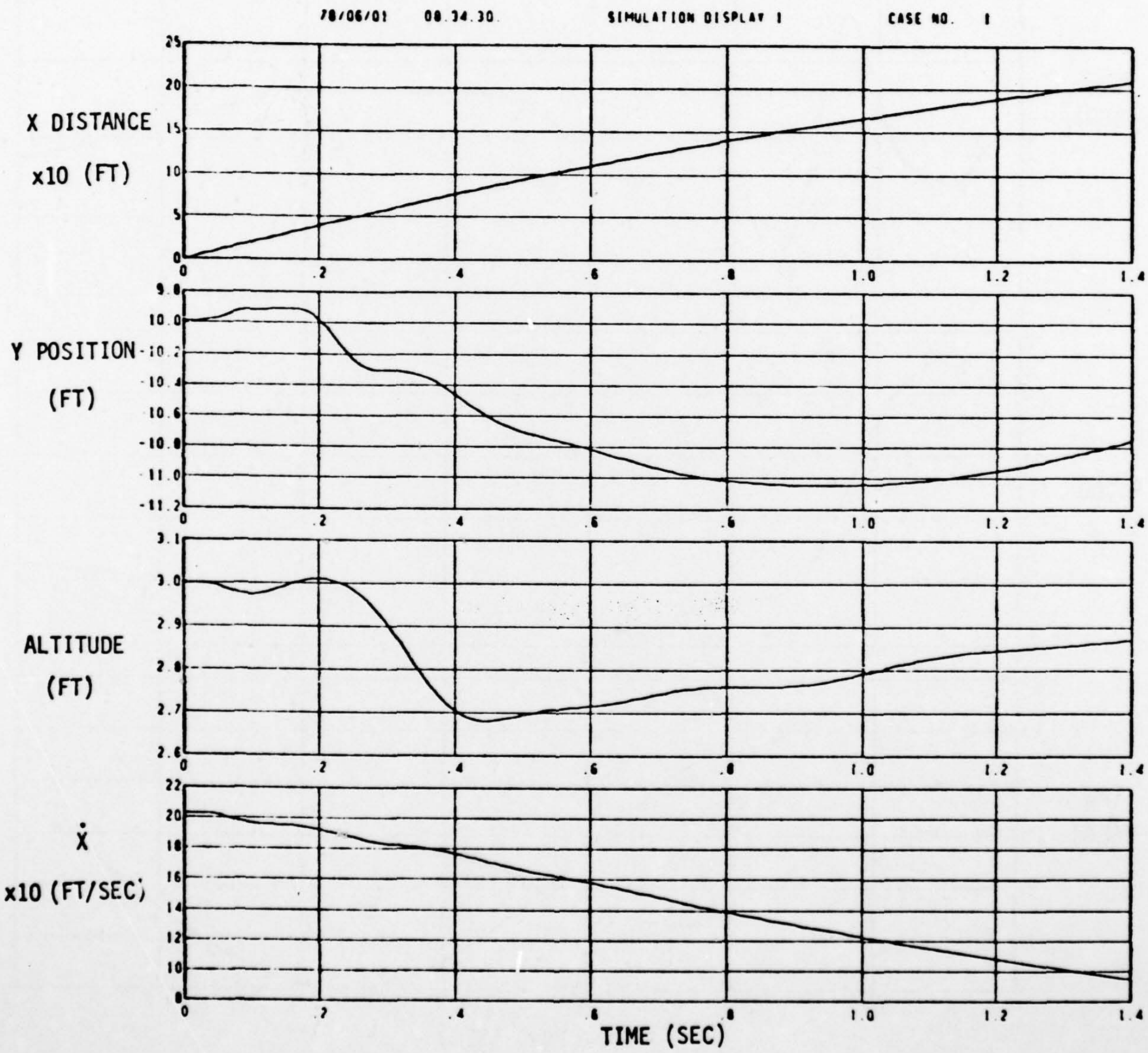


Figure 129 Time History Plots for Arresting System, DISPLAY1

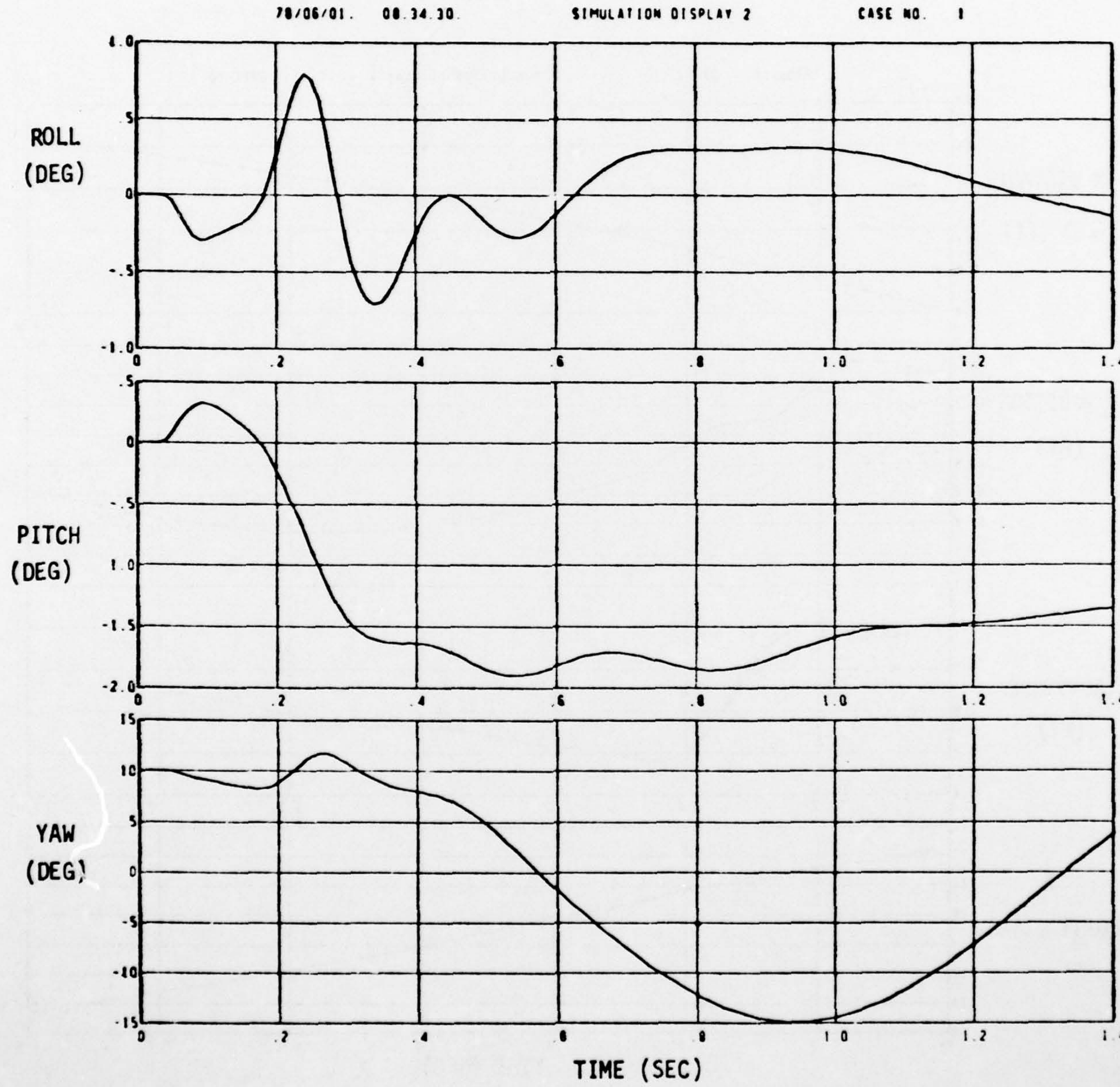


Figure 130 Time History Plots for Arresting System, DISPLAY2

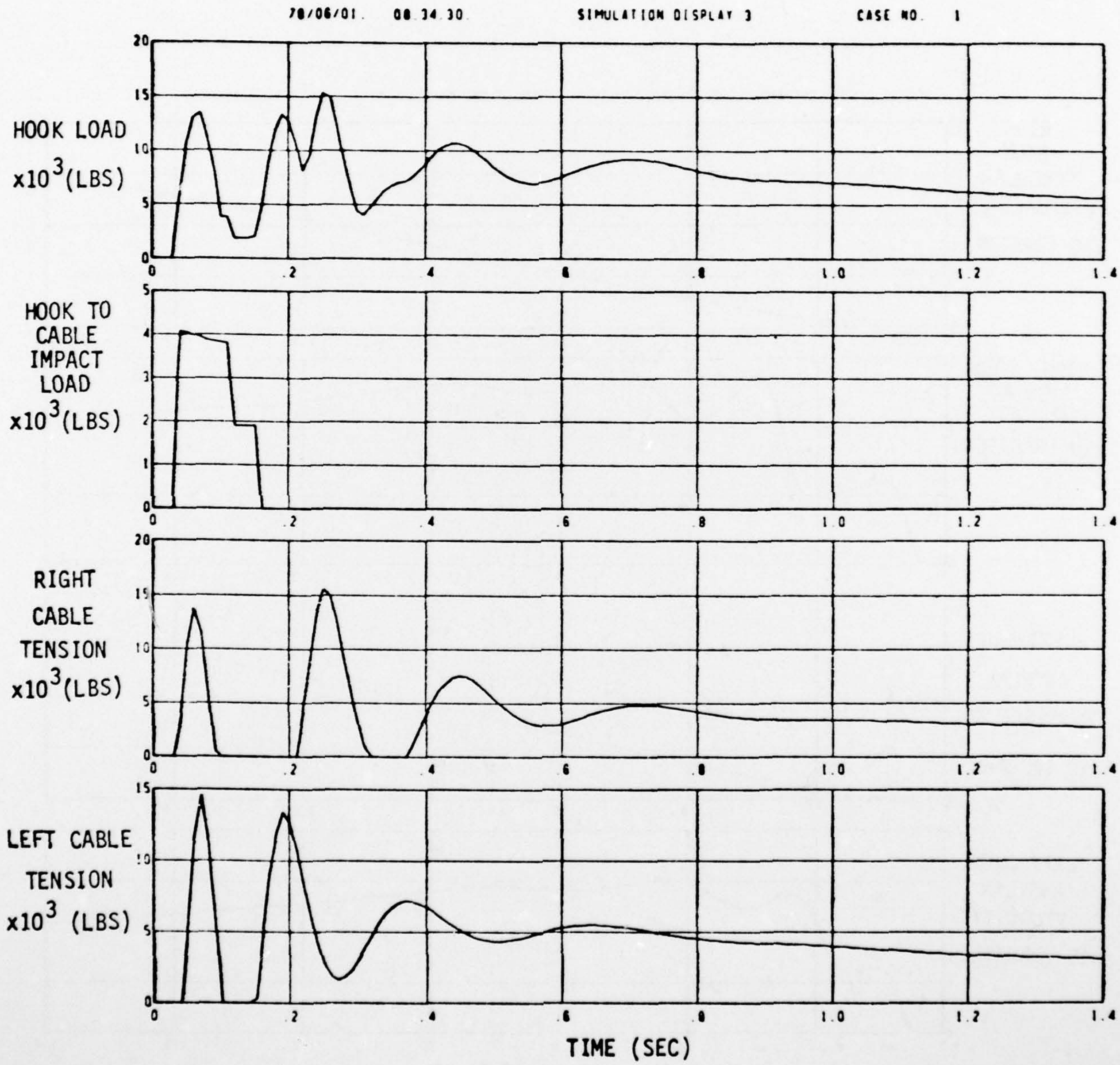


Figure 131 Time History Plots for Arresting System, DISPLAY3

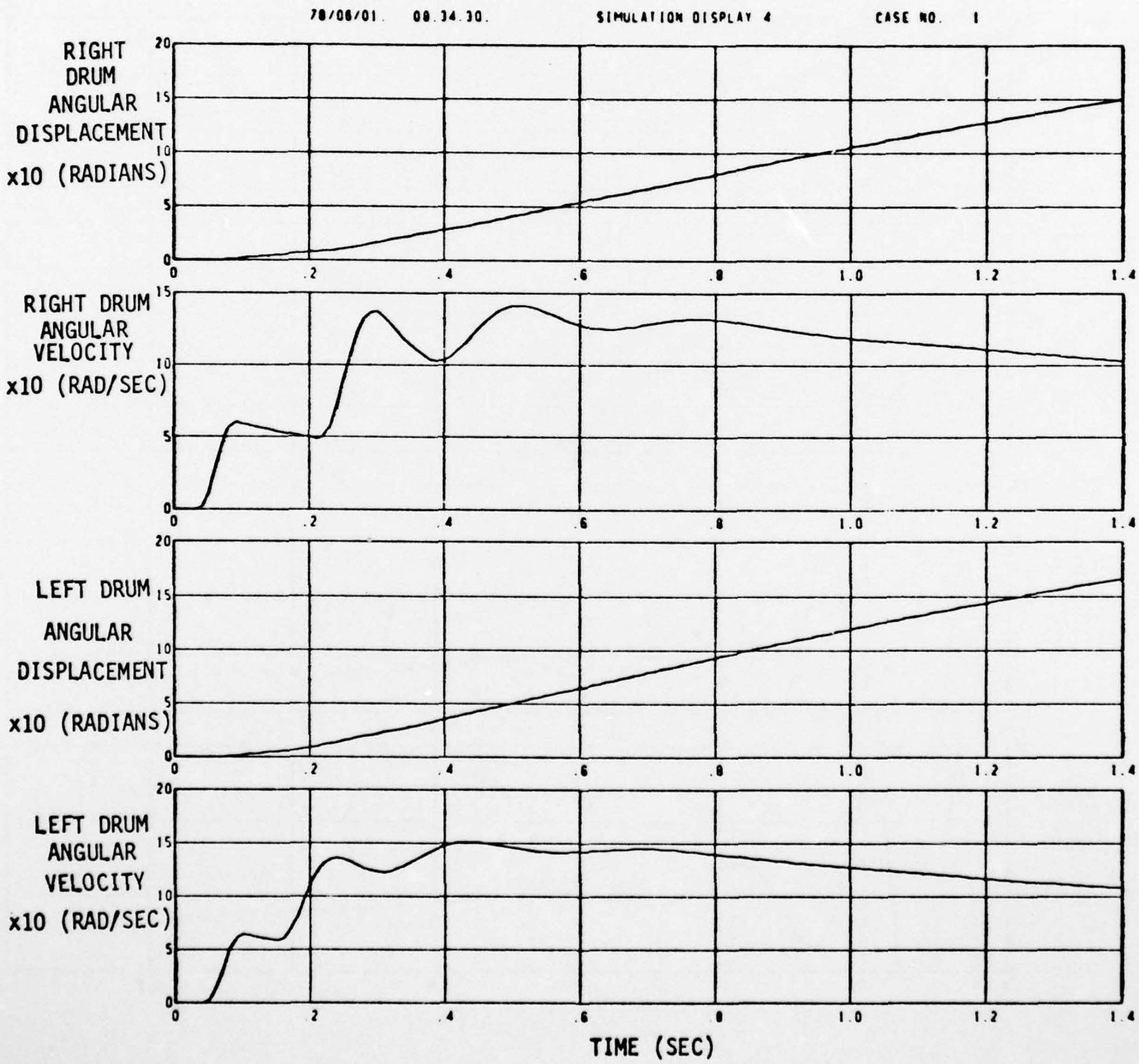


Figure 132 Time History Plots for Arresting System, DISPLAY4

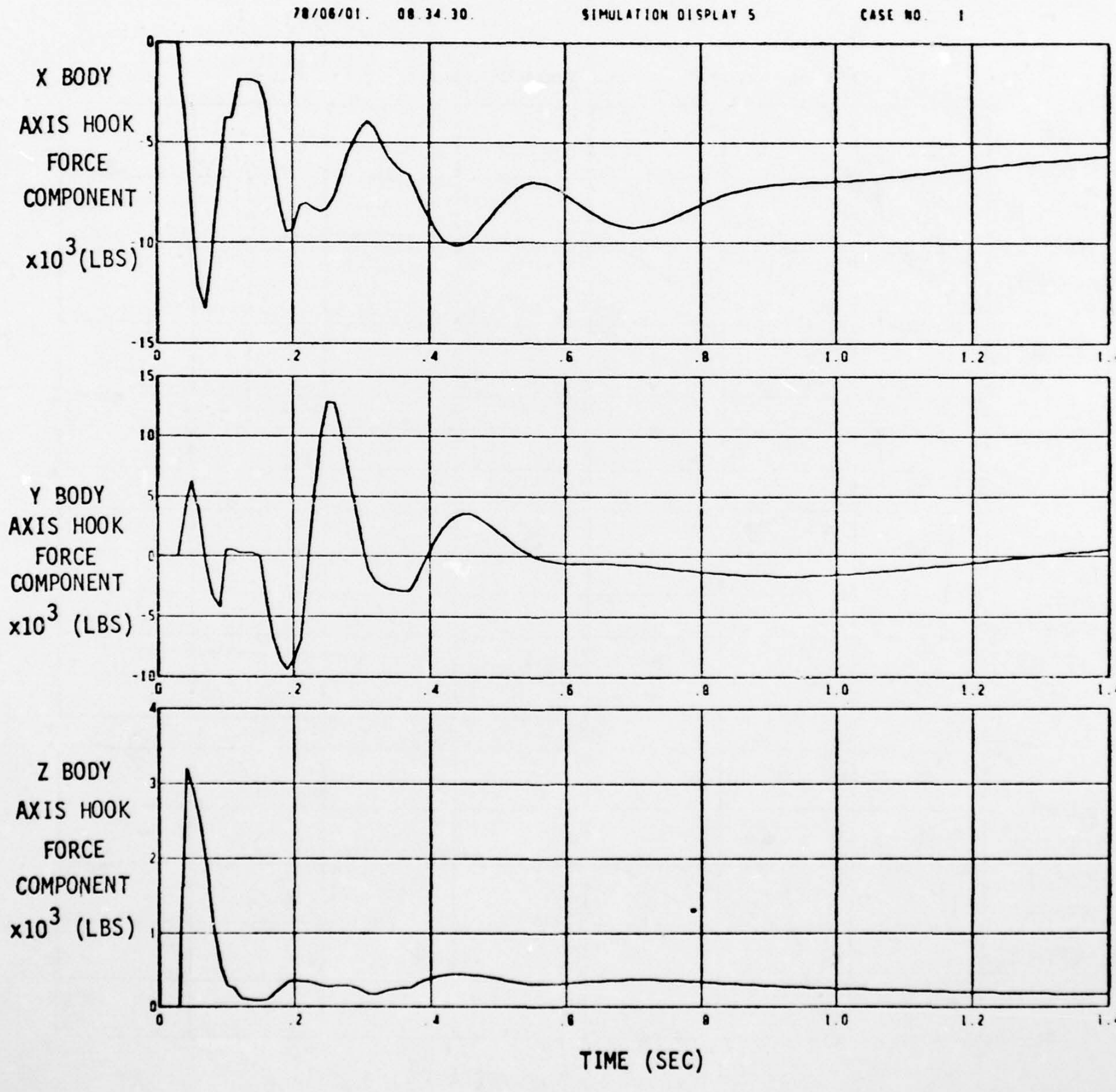


Figure 133 Time History Plots for Arresting System, DISPLAY5

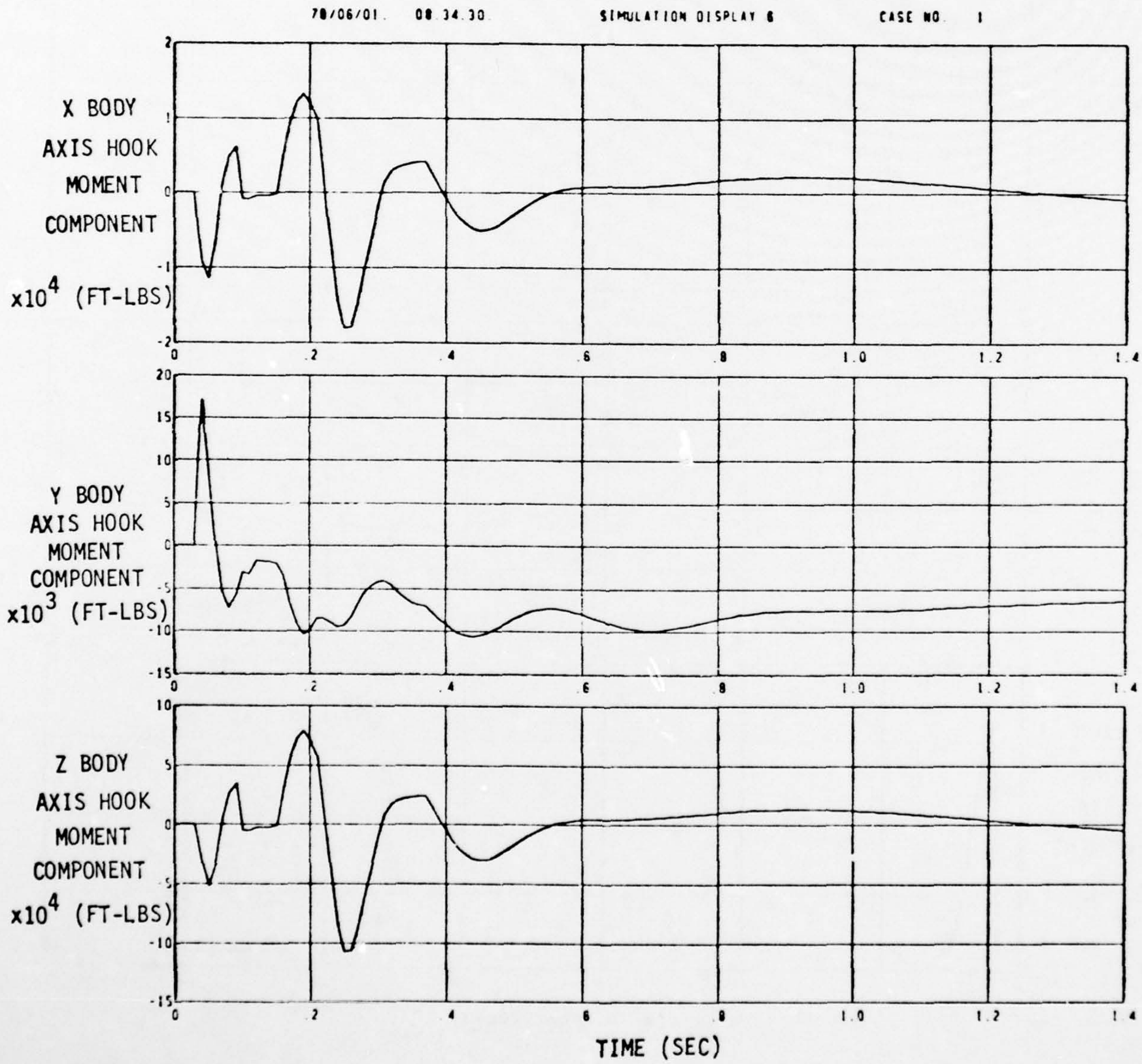


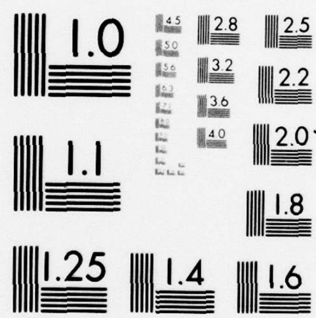
Figure 134 Time History Plots for Arresting System, DISPLAY6

AD-A080 489 BOEING AEROSPACE CO SEATTLE WA BOEING MILITARY AIRPL--ETC F/6 9/2
EASY ACLS DYNAMIC ANALYSIS. VOLUME III. DESCRIPTION OF SIMULATI--ETC(U)
SEP 79 M K WAHI, P R PERKINS, G S DULEBA F33615-77-C-3054

UNCLASSIFIED AFFDL-TR-79-3105-VOL-3 NL

3 OF 4

AD
A080489



MICROCOPY RESOLUTION TEST CHART
NATIONAL BUREAU OF STANDARDS-1963-A

The spikes in the tension load time histories during the first tenth of a second are caused by over speeding of the tape drums (see Figure 132). Smoother tension time histories can be achieved if compatible arresting system parameters are determined for a specified vehicle gross weight. A refinement of arresting system parameters was not considered necessary for the component checkout procedure.

SECTION VIII

XC-8A AIRCRAFT SIMULATION

8.1 Objectives

The main objectives of this task were to develope a 6-DOF rigid body model of the XC-8A aircraft and to verify that model by comparing the model simulation results with the Air Force XC-8A Flight Simulator results for the same flight conditions. In addition to the conventional aerodynamic terms, the model must include the aerodynamic effects of a deflated trunk, two ASP-10s and two wing skids.

To validate the EASY XC-8A model, a longitudinal and a lateral stability evaluation was chosen. During these two evaluations the aircraft was perturbed from an initial flight configuration by a saw tooth shaped elevator or rudder deflection. A complete listing of the initial flight configuration prior to the rudder or elevator deflections is presented in Table 9.

8.2 Technical Approach

8.2.1 Model File

Five standard EASY components along with several pages of Fortran coding were assembled to form the XC-8A 6-DOF rigid body aerodynamic model. In general terms this model can be described as a nonlinear, variable coefficient model with ground effects. Both the coefficients and the ground effects are determined from the aircraft angle of attack, total coefficient of thrust and flap deflection.

The EASY XC-8A model file and the computer generated XC-8A model schematic are given in Figures 135 and 136. The EASY standard components VA, OL, DL and SG in the model file form the 6-DOF rigid body aerodynamic model. Component TB is used to input the elevator and rudder schedules required during the longitudinal and lateral stability evaluations.

TABLE 9 XC-8A FLIGHT CONFIGURATION

(SIMULATION TIME = 0 SEC)

FLAPS	0.0 DEG
TRUE AIRSPEED	253.0 FT/SEC
ALTITUDE(c.g.)	5000 FT
GROSS WEIGHT	41000 LBS
CENTER OF GRAVITY (c.g.)	40% MAC
FSCG	358.2 IN
WLCG	160.0 IN
MOMENTS OF INERTIA:	
IXX	301300 SLUGS-FT^2
IYY	267000 SLUGS-FT^2
IZZ	508600 SLUGS-FT^2
IXZ	24190 SLUGS-FT^2
THRUST	0.0 LBS
ATTITUDE:	
ANGLE OF ATTACK (α)	3.39 DEG
SIDESLIP ANGLE (β)	0.0 DEG
FLIGHT PATH ANGLE (γ)	0.0 DEG
PITCH (θ)	3.39 DEG
ROLL (ϕ)	0.0 DEG
YAW (ψ)	0.0 DEG
LONGITUDINAL CONTROL SURFACE DEFLECTIONS:	
ELEVATOR	-.037 DEG
SPOILER	0.0 DEG
LATERAL CONTROL SURFACE DEFLECTIONS:	
RUDDER	0.0 DEG
AILERON	0.0 DEG
STATE RATES WHICH HAVE NONZERO VALUES:	
$\dot{u}$	-2.30 FT/SEC^2
$\dot{w}$	-.13 FT/SEC^2
$\dot{x}$	253 FT/SEC

```
MODEL DESCRIPTION    XC-8A AIRCRAFT SIMULATION (TASK 3)
FORTRAN STATEMENTS
      REAL L1,M1,N1
ADD PARAMETERS=THETA
FSENG,BLENG,WLENG,FSCG,BLCG,WLCG
TRUNK,CGRAV,FLAPS,SPOIL
ROOTC,ASPECT,SH,FS25CR,WL25CR
FS75CR,WL75CR,FSTAIL,WLTAIL
THRRT,THRLT
ADD VARIABLES
GAMMA
CTRT,CTLT,CTTOT,FFBETA
DELCLGR,DELCDLG,DELCMG
ADD TABLES
TBCLO,46,TBCL20,46,TBCL40,46
TBCLH,38,TBX,-25,TBLLO,54,TBDEL,-13
LOCATION=58    VA    INPUTS=SG
LOCATION=74    TB
FORTRAN STATEMENTS
      GAMMA=PITSG-AL VA
      ELEOL=A2 TB
      RUDDL=B2 TB
C*********************************************************************
C*********************************************************************
C  THESE FORTRAN STATEMENTS UPDATE THE COS(ALPHA) AND SIN(ALPHA)
C  WITHOUT CHANGING THE VALUE OF ALSVA. THIS PROVIDES FOR CORRECT
C  RESOLUTION OF THE LIFT AND DRAG FORCES FROM THE STABILITY
C  AXIS INTO THE BODY AXIS.
C  CAUTION- THESE FORTRAN STATEMENTS CANNOT BE USED WHEN THE
C  *******  AERODYNAMIC DERIVATIVES ARE REFERENCED TO THE BODY AXIS.   1
      RPD=.01745329
      SALVA=SIN(AL VA*RPD)
      CALVA=COS(AL VA*RPD)
      PO VA=P  SG*CALVA+R  SG*SALVA
      RO VA=R  SG*CALVA-P  SG*SALVA
C*********************************************************************
C*********************************************************************
C THE FOLLOWING IF STATEMENTS LIMIT THE ALLOWABLE DEFLECTIONS
C OF THE ELEVATOR,RUDDER,FLAPS AND AILERION WHEEL. THEY ALSO
C LIMIT THE ENGINE THRUST TO A SPECIFIED RANGE.
C*********RANGE OF ELEVATOR DEFLECTION= -25.0 TO +15.0 DEG.
C          (A POSITIVE ELEVATOR DEFLECTION IS DOWN)
      IF(ELEOL.LT.-25.0) ELEOL=-25.0
      IF(ELEOL.GT.15.0) ELEOL=15.0                                     2
C*********RANGE OF RUDDER DEFLECTION= -20.0 TO +20.0 DEG.
C          (A POSITIVE RUDDER DEFLECTION IS TO THE PILOTS LEFT)
      IF(RUDDL.LT.-20.0) RUDDL=-20.0
      IF(RUDDL.GT.20.0) RUDDL=20.0
C*********RANGE OF FLAP DEFLECTION= 0.0 TO +40.0 DEG.
```

Figure 135 XC-8A Aircraft Simulation Model File

```
      IF(FLAPS.LT.0.0) FLAPS=0.0
      IF(FLAPS.GT.40.0) FLAPS=40.0
C*********RANGE OF AILERION WHEEL ROTATION= -75.0 TO +75.0 DEG.
C          (A POSITIVE AILERION DEFLECTION TIPS THE RIGHT WING DOWN)
      IF(AILDL.LT.-75.0) AILDL=-75.0
      IF(AILDL.GT.75.0) AILDL=75.0
C*********THE MAXIMIUM THRUST PER ENGINE=+9000.0 LBS..
      IF(THRRT.GT.9000.0) THRRT=9000.0
      IF(THRLT.GT.9000.0) THRLT=9000.0
C*********THE MAXIMIUM REVERSE THRUST PER ENGINE=-9000.0
      IF(THRRT.LT.-9000.0) THRRT=-9000.0
      IF(THRLT.LT.-9000.0) THRLT=-9000.0
C*********************************************************************
C*********************************************************************
C  THESE FORTRAN STATEMENTS CALCULATE THE BODY AXIS FORCES
C  AND MOMENTS PRODUCED BY THE ENGINE THRUST. THE THRUST
C  LOCATION AND ANGLE OF THRUST VECTOR MUST BE SPECIFIED.
C  THE MODEL CAN ACCEPT REVERSE THRUST AND/OR DIFFERENTIAL
C  THRUST SCHEDULES.
C*********THRLT=LEFT ENGINE THRUST(LBS.)
C*********THRRT=RIGHT ENGINE THRUST(LBS.)
C*********XENG,YENG AND ZENG=THE XYZ BODY AXIS MOMENT ARM
C         VECTORS (FEET)
C  THE REQUIRED INPUT PARAMETERS ARE
C*********THETA=THE ANGLE OF THRUST APPLICATION IN THE XZ PLANE
C            (DEGREES)
C*********FSENG,BLENG,WLENG. THESE ARE THE THE FUSELAGE
C         STATION,BUTT LINE, AND WATER LINE LOCATIONS
C         OF THE RIGHT ENGINE THRUST VECTOR. (IN.)
C*********FSCG,BLCG,WLCG. THESE ARE THE FUSELAGE STATION,
C         BUTT LINE AND WATER LINE LOCATIONS OF THE C.G..(IN.)
      RPD=.01745329
      FLPRAD=FLAPS*RPD
      THRAD=THETA*RPD
      COSINE=COS(THRAD)
      SINE=SIN(THRAD)
      XENG=(FSCG-FSENG)/12.0
      YENG=(BLENG-BLCG)/12.0
      ZENG=(WLCG-WLENG)/12.0
      THDIF=THRLT-THRRT
      THRT=THRRT
      THLT=THRLT
      IF(THRT.LT.0.0) THRT=0.0
      IF(THLT.LT.0.0) THLT=0.0
      DIF=THLT-THRT
      FX1OL=(THRLT+THRRT)*COSINE
      FY1DL=(.002+.0183*FLPRAD)*DIF
      FZ1OL=-(THRLT+THRRT)*SINE
      T1=-.0176*(THRLT+THRRT)
```

2

3

Figure 135 XC-8A Aircraft Simulation Model File (Continued)

```
      T2=(.0125+.1095*FLPRAD)*DIF
      TX1DL=T1+T2                                                          3
      TY1OL=FX1OL*ZENG-FZ1OL*XENG
      TZ1DL=(.1575-.0322*FLPRAD)*THDIF
C*********************************************************************
C*********************************************************************
C THESE FORTRAN STATEMENTS DETERMINE THE AERODYNAMIC
C DERIVATIVES AND COEFFICIENTS USED IN COMPONENTS OL AND DL.
C THE CALCULATED DERIVATIVES AND COEFFICIENTS ARE ALL WITH
C RESPECT TO THE STABILITY AXIS AND ARE ALL NONDIMENSIONAL
C VALUES. THE MOMENT TERMS ARE FOR A C.G. LOCATION AT 40
C PERCENT MAC. XP1OL CORRECTS THE DATA FOR A DIFFERENT
C C.G. LOCATION.
C THE XC-8A AERODYNAMIC MODEL WAS OBTAINED FROM G.KURYLOWICH
C AIR FORCE FLIGHT DYNAMICS LABORATORY.
C         *********************************************
C         * THE FOLLOWING VARIABLES AND PARAMETERS    *
C         * ARE NEEDED TO DETERMINE THE AERODYNAMIC   *
C         * COEFFICIENTS AND DERIVATIVES. THESE ARE   *
C         * NOT EASY STANDARD COMPONENT VARIABLES OR  *
C         * PARAMETERS.                               *
C         *   TRUNK-  THE TRUNK IS DEFLATED WHEN      *
C         *           TRUNK=0.0, AND FULLY INFLATED   *
C         *           WHEN TRUNK=1.0.                 *
C         *   CGRAV-  THE LOCATION OF THE C.G.. IT    *
C         *           IS  INPUT AS A PERCENT MAC.     *
C         *   FLAPS-  THE ANGULAR DEFLECTION OF THE   *
C         *           FLAPS,INPUT IN DEGREES.         *
C         *   SPOIL-  THE FRACTION OF THE TOTAL       *            4
C         *           SPOILER DEFLECTION. THE RANGE   *
C         *           OF SPOIL IS FROM 0.0 TO 1.0. THE *
C         *           SPOILERS ARE FULLY DEPLOYED WHEN *
C         *           SPOIL=1.0, AND FULLY RETRACTED  *
C         *           WHEN SPOIL=0.0.                 *
C         *********************************************
C THIS NEXT GROUPING OF FORTRAN STATEMENTS EVALUATES
C FREQUENTLY USED EXPRESSIONS PRESENT IN THE AERODYNAMIC
C EQUATIONS.
C*********CTRT IS THE RIGHT ENGINE COEFFICIENT OF THRUST
C*********CTLT IS THE LEFT ENGINE COEFFICIENT OF THRUST.
C*********CTTOT IS THE TOTAL COEFFICIENT OF THRUST.
      CTRT=THRRT/QS VA
      CTLT=THRLT/QS VA
      CTTOT=CTRT+CTLT
C*********IF CTTOT IS NEGATIVE,CTTOT IS THEN SET TO ZERO.
      IF(CTTOT.LT.0.0) CTTOT=0.0
      CTT2=CTTOT**2.
      RPD=.01745329
      ALRAD=AL VA*RPD
```

Figure 135 XC-8A Aircraft Simulation Model File (Continued)

```
      BERAD=BE VA*RPD
      COS2BE=(COS(BERAD))**2.
      SIN2BE=(SIN(BERAD))**2.
      BEABS=ABS(BERAD)
      FFBETA=ABS(1.-.955*(BEABS-.523))
      IF(BEABS.LE..523) FFBETA=1.0
      FLPRAD=FLAPS*RPD
      FLP2=FLPRAD**2.
      ABZF=(WLCG-160.)/12.0
      ABZC=(WLCG-74.6)/12.0
      PIH=3.141592654/2.0
C AERODYNAMIC CONTRIBUTIONS FOR THE APUS,DIFFUSERS AND CUSHION.
C ( A 40 PERCENT C.G. LOCATION WAS USED FOR ALL TERMS.)
      ABCL=1.59/QC VA
      IF(ALTSG.LE.11.0) ABCL=0.0
      ABSB=ABS(SIN(BE VA))
      CDMON=5.19/VT VA
C CUSHION RETRACTED INCREMENTS OGE FANS CLOSED
      RETCD=.004
      RETCY=0.0
      RETCM=-.004*(ABZF-7.9*ALRAD)/C  OL
      RETCLL=0.0
      RETCN=0.0
C CUSHION EXTENDED INCREMENTS FAN CLOSED
      EXTCD=.02
      EXTCY=-.05*ABSB
      EXTCM=-.02*(ABZC-5.54*ALRAD)/C  OL                              4
      EXTCLL=-EXTCY*ABZC/B  DL
      EXTCN=EXTCY*5.54/B  DL
C CUSHION EXTENDED INCREMENTS WITH FANS OPERATIONAL
EMCD=CDMON+ABCL*ALRAD
      EMCY=-CDMON
      EMCM=-CDMON*(ABZF-5.6*ALRAD)/C  OL+ABCL*5.54/C  OL
      EMCLL=-EMCY*ABZF/B  DL
      EMCN=EMCY*5.6/B  DL
C * * * * * * * * * * * * * * * * * * * * * * * * *
C
C AERODYNAMIC LIFT DERIVATIVES AND COEFFICIENTS
C
C********** BASIC LIFT COEFFICIENT **********
      Z1=(.35+2.75*FLPRAD-.616*FLP2+(.15+1.865*FLPRAD
     1-.821*FLP2)*CTTOT+(5.27+.72*CTTOT)*ALRAD)
     2*COS2BE
      ZO OL=-Z1
C********** ELEVATOR DERIVATIVE **********
      ZDEOL=-.566*COS2BE
C********** SPOILER DERIVATIVE **********
      ZSPOL=-(-.25-.573*FLPRAD)
      SPOOL=SPOIL/RPD
```

Figure 135 XC-8A Aircraft Simulation Model File (Continued)

```
C        SPOIL IS DIVIDED BY RPD TO NEGATE THE EFFECTS
C        OF THE AUTOMATIC DEGREE TO RADIAN CONVERSION.
C********** TRUNK COEFFICIENT **********
      ZTROL=-TRUNK*ABCL
C * * * * * * * * * * * * * * * * * * * * * * * * *
C
C AERODYNAMIC DRAG DERIVATIVES AND COEFFICIENTS
C
C********** BASIC DRAG COEFFICIENT **********
      CL2=(ZO OL+ZDEOL*ELEOL*RPD+ZTROL+ZSPOL*SPOOL*RPD)**2.
      X1=(.026+.0394*CL2+(.115+.46*CTTOT-.12*CTT2
     1)*FLPRAD)*COS2BE
      IF(X1.LT.0.0) X1=0.0
C          MODIFICATION OF BASIC DRAG COEFFICIENT
C          WHEN VT VA IS LESS THAN 70.0 FPS.
      X2=0.0
      IF(VT VA.GT.70.0) GO TO 15
      X2=(.229*FLPRAD+.176*FLP2)*CTTOT
      B=1.0-(VT VA-50.0)/20.0
      IF(50.0.LE.VT VA.AND.VT VA.LE.70.0) X2=X2*B
  15  CONTINUE
      XO OL=-X1-X2
C********** TRUNK COEFFICIENT **********
      XTROL=-(RETCD+TRUNK*(EXTCD+EMCD))
C********** SPOILER DERIVATIVE **********
      XSPOL=(-.0125-(.1*FLPRAD))
C * * * * * * * * * * * * * * * * * * * * * * * * *
C
C AERODYNAMIC SIDEFORCE DERIVATIVES AND COEFFICIENTS
C
C********** BETA DERIVATIVE **********
      IF(BEABS.GT..436) GO TO 18
      A=-.8-.515*FLPRAD
      B=-.1-.702*FLPRAD
      C=-.57-1.89*FLPRAD+.82*FLP2
      D=.573-1.64*FLPRAD
      IF(FLPRAD.GE..349) D=-1.145
      Y1=A+B*CTTOT+(C+D*CTTOT)*ALRAD
      GO TO 19
  18  A=.08+.225*FLPRAD
      B=(.025+.322*FLPRAD)*CTTOT-.079*BERAD
      IF(ABS(CTTOT).LT..1) B=B+.079*BERAD
      IF(BERAD-PIH) 500,501,501
 500  Y1=-(1.0/BEABS)*(1.38*SIN2BE+A*(1.0
     1-(BEABS-.436)/1.134)+B)
      GO TO 19
 501  Y1=-(1.0/BEABS)*(1.38*SIN2BE+A*(PIH-BEABS)/
     1   1.134+B)
  19  YB DL=Y1
```
4

Figure 135 XC-8A Aircraft Simulation Model File (Continued)

```
C********** AILERION DERIVATIVE **********
      YDADL=.0277*(.0473+.0545*FLPRAD)*FFBETA
C********** RUDDER DERIVATIVE AND EFFECTIVENESS **********
      YDRDL=.407
      YBRDL=FFBETA
C********** ROLL RATE DERIVATIVE **********
      YP DL=-.06
C********** YAW RATE DERIVATIVE **********
      YR DL=.704
C********** TRUNK COEFFICIENT **********
      YTRDL=RETCY+TRUNK*(EXTCY+EMCY)
C * * * * * * * * * * * * * * * * * * * * * * * *
C
C AERODYNAMIC PITCHING MOMENT DERIVATIVES AND COEFFICIENTS
C
C********** BASIC PITCHING MOMENT COEFFICIENT **********
      IF(CTTOT.GE.1.0) GO TO 20
      A=.2-.2*CTTOT
      B=.251-.036*CTTOT
      C=-.245*CTTOT
      D=-3.69+3.616*CTTOT
      E=-1.645-5.375*CTTOT
      F=4.71*CTTOT
      GO TO 21
   20 A=.2*(CTTOT-1.0)
      B=.215+1.002*(1.0-CTTOT)
      C=-.49+.245*CTTOT
      D=-.074+1.566*(1.0-CTTOT)
      E=-14.04+7.02*CTTOT
      F=9.42-4.71*CTTOT
   21 M1=A+B*FLPRAD+C*FLP2+(D+E*FLPRAD+F*FLP2)
     1*(ALRAD+.1745)**2.
      MO OL=M1
C********** PITCH RATE DERIVATIVE **********
      MQ OL=-38.4
C********** ALPHA DOT DERIVATIVE **********
      MADOL=-7.-5.*CTTOT
      IF(CTTOT.GE.1.0) MADOL=-12.0
C********** TRUNK COEFFICIENT **********
      MTROL=(RETCM+EXTCM+EMCM)*TRUNK
C********** ELEVATOR DERIVATIVE **********
      MDEOL=-2.43*COS2BE
C********** SPOILER DERIVATIVE
      MSPOL=(.02+.1145*FLPRAD)
C * * * * * * * * * * * * * * * * * * * * * * * *
C
C AERODYNAMIC ROLL MOMENT DERIVATIVES AND COEFFICIENTS
C
C********** BETA DERIVATIVE **********
```

4

Figure 135 XC-8A Aircraft Simulation Model File (Continued)

```
      IF(BEABS.GT..436) GO TO 31
      L1=(-.164+.043*FLPRAD+(.014+.1575*FLPRAD-.0924*FLP2)
     1*CTTOT+(.172+(.1035-.0186*FLPRAD-.082*FLP2)
     2*CTTOT)*ALRAD)
      GO TO 32
   31 A=.01875*FLPRAD
      B=(.0041+.0326*FLPRAD)*CTTOT
      IF(BEABS-PIH) 510,511,511
  510 L1=(1.0/BEABS)*(-.022-.235*SIN2BE*COS2BE+
     1A*(1.0-(BEABS-.436)/1.134)+B)
      GO TO 32
  511 L1=(1.0/BEABS)*(-.022-.235*SIN2BE*COS2BE+A
     1   *(PIH-BEABS)/1.134+B)
   32 LB DL=L1
C********** ROLL RATE DERIVATIVE **********
      LP DL=-.53-.1*FLPRAD+(.08-.0572*FLPRAD)*CTTOT
     1+.516*ALRAD
C**********YAW RATE DERIVATIVE **********
      LR DL=.15+.587*FLPRAD+.975*ALRAD
C********** AILERION DERIVATIVE **********
      LDADL=(.0473+.0545*FLPRAD)*FFBETA
C********** RUDDER DERIVATIVE AND EFFECTIVENESS PARAMETER **********
      LDRDL=.184*SIN(.265-ALRAD)
      LBRDL=FFBETA
C********** TRUNK COEFFICIENT **********
      LTRDL=RETCLL+TRUNK*(EXTCLL+EMCLL)                                  4
C * * * * * * * * * * * * * * * * * * * * * * * * *
C
C AERODYNAMIC YAW MOMENTS DERIVATIVES AND COEFFICIENTS
C
C********** BETA DERIVATIVE **********
      IF(BEABS.GT..436) GO TO 37
      A=.125+.0329*FLPRAD
      B=-.015-.0415*FLPRAD
      C=.057+.52*FLPRAD-.495*FLP2
      D=-.0573-.181*FLPRAD
      IF(FLPRAD.GE..349) D=-.1204+.1315*(FLPRAD-.349)
      N1=A+B*CTTOT+(C+D*CTTOT)*ALRAD
      GO TO 38
   37 A=.02475+.01435*FLPRAD
      B=(.006+.0236*FLPRAD)*CTTOT
      IF(BEABS-PIH) 520,521,521
  520 N1=(1.0/BEABS)*(.106*SIN2BE+A*(1.0
     1-(BEABS-.436)/1.134)-B)
      GO TO 38
  521 N1=(1.0/BEABS)*(.106*SIN2BE+A*(PIH-BEABS)/1.134-B)
   38 NB DL=N1
C********** YAW RATE DERIVATIVE **********
      NR DL=-.22-.06*FLPRAD-(.129+.308*FLPRAD)*ALRAD
```

Figure 135 XC-8A Aircraft Simulation Model File (Continued)

```
C********** ROLL RATE DERIVATIVE **********
      NP DL=((-.1+.009*CTTOT)*FLPRAD)/.699+
     1(-.5-.145*CTTOT*(FLPRAD-.699)/.699)*ALRAD
C********** AILERION DERIVATIVE **********
      NDADL=-(.0467+.704*ALRAD)*LDADL
C********** RUDDER DERIVATIVE AND EFFECTIVENESS PARAMETER **********
      NDRDL=-.184*COS(.265-ALRAD)                                          4
      NBRDL=FFBETA
C********** TRUNK COEFFICIENT **********
      NTRDL=RETCN+TRUNK*(EXTCN+EMCN)
C XP1OL CORRECTS THE AERODYNAMIC DATA FOR A C.G. LOCATION
C DIFFERENT FROM 40 PERCENT MAC USED IN DERIVING THE DATA.
      XP1OL=C  OL*(.01*CGRAV-.4)
C*********************************************************************
C*********************************************************************
C  THE FOLLOWING FORTRAN STATEMENTS EVALUATE THE GROUND EFFECTS
C  FOR THE AERODYNAMIC MODEL. THE PROCEEDURE USED TO EVALUATE
C  GROUND EFFECTS WAS TAKEN FROM DATCOM SECTION 4.7 METHOD 1.
C  THE REQUIRED INPUT PARAMETERS ARE
C*********ROOTC=ROOT CHORD (CR) LENGTH (FEET)
C*********SH=HORIZONTAL STABILIZER AREA (SQUARE FEET)
C*********ASPECT=ASPECT RATIO
C*********FS25CR,WL25CR=THE FUSELAGE STATION AND WATER
C         LINE LOCATIONS OF .25 ROOT CHORD.
C*********FS75CR,WL75CR=THE FUSELAGE STATION AND WATER
C         LINE LOCATIONS OF THE .75 ROOT CHORD.
C*********FSTAIL,WLTAIL=FUSELAGE STATION AND WATER
C         LINE LOCATIONS OF THE HORIZONTAL STABILIZERS
C         AERODYNAMIC CENTER.
C  THE REQUIRED INPUT TABLES ARE
C*********TBCL0,TBCL20,TBCL40= THE TOTAL AIRPLANES COEFFICIENT
C         OF LIFT AT 0.0,20.0,40.0 DEG. FLAPS RESPECTIVELY.
C         THESE TABLES ARE FUNCTIONS OF ALPHA AND CTTOT.                   5
C*********TBCLH=THE HORIZONTAL STABILIZER COEFFICIENTS OF
C         INPUT AS FUNCTIONS OF HORIZONTAL ALPHA AND ELEOL.
C*********TBX=THE X VARIABLES FROM DATCOM FIGURE 4.7.1-14.
C*********TBLLO=THE L/LO-1. VARIABLE FROM DATCOM FIGURE 4.7.1-15.
C*********TBDEL=THE DELTA DELTA CL VARIABLE FROM DATCOM
C         FIGURE 4.7.1-17 FOR SLOTTED FLAPS.
C  ASSUMPTIONS
C         1. SWEEP ANGLE OF THE QUARTER CHORD LINE
C            OF THE WING = 0.0 DEG..
C         2. THE DYNAMIC PRESSURE OF THE HORIZONTAL
C            STABILIZER EQUALS THE DYNAMIC PRESSURE
C            AT INFINITY.
C         3. THE WINGS ARE STRAIGHT AND PERPENDICULAR
C            TO THE X-Z PLANE. THUS H.75CR=H.75B/2.
C            SEE DATCOM FIGURE 4.7.1-14.
C         4. THE MAIN WINGS AERODYNAMIC CENTER IS ASSUMED
```

Figure 135 XC-8A Aircraft Simulation Model File (Continued)

```
C           TO BE AT .25C..
C  IF THE AIRCRAFTS ALTITUDE IS GREATER THAN TWICE THE
C  WING SPAN THE GROUND EFFECTS ARE SKIPPED.
      DELCLGR=0.0
      DELCDLG=0.0
      DELCMG=0.0
      IF(ALTSG.GT.2.*B  DL) GO TO 777
C  THE NEXT GROUPING OF FORTRAN STATEMENTS EVALUATES
C  FREQUENTLY USED VARIABLES IN THE GROUND EFFECTS
C  CALCULATIONS.
C
      SRATIO=SH/S  VA
      EPSILON=AL VA*(.24-.0015*FLAPS)
      TAILA=AL VA-EPSILON
      X75CR=(FSCG-FS75CR)/12.0
      Z75CR=(WLCG-WL75CR)/12.0
      X25CR=(FSCG-FS25CR)/12.0
      Z25CR=(WLCG-WL25CR)/12.0
      XTAIL=(FSCG-FSTAIL)/12.0
      ZTAIL=(WLCG-WLTAIL)/12.0
      RPD=.01745329
      SP=SIN(PITSG*RPD)
      CR=COS(ROLSG*RPD)
      CP=COS(PITSG*RPD)
      CRCP=CR*CP
      H75CR=X75CR*SP-Z75CR*CRCP+ALTSG
      IF(H75CR.LE.11.25) GO TO 777
      H25CR=X25CR*SP-Z25CR*CRCP+ALTSG
      HTAIL=XTAIL*SP-ZTAIL*CRCP+ALTSG
      H75=2.*H75CR/B  DL
      H25=H25CR/ROOTC
      CRB=ROOTC/B  DL
C  THE NEXT GROUPING OF FORTRAN STATEMENTS EVALUATES
C  THE TOTAL AIRCRAFTS COEFFICIENT OF LIFT BY TABLE
C  LOOK UP ROUTINES INPUT AS FUNCTIONS OF ALPHA AND CTTOT.
C  THERE ARE 3 INPUT TABLES FOR FLAP SETTINGS OF 0.0,20.0
C  AND 40.0 DEGREES. A LINEAR INTERPELATION IS MADE
C  FOR FLAP SETTINGS BETWEEN THESE VALUES.
      I=0
      IF(FLAPS.GE.20.) GO TO 2
      T=FLAPS/20.0
      GO TO 4
    2 T=(FLAPS-20.)/20.
    4 A=AL VA
    5 I=I+1
      IF(FLAPS.GE.20.) GO TO 14
      COEL1=TBLU2(A,CTTOT,TBCL0(7),TBCL0(4),TBCL0(17)
     1,1,1,10,3,10,3)
      COEL2=TBLU2(A,CTTOT,TBCL20(7),TBCL20(4),TBCL20(17)
```
5

Figure 135 XC-8A Aircraft Simulation Model File (Continued)

```
      1,1,1,10,3,10,3)
       GO TO 150
   14  COEL1=TBLU2(A,CTTOT,TBCL20(7),TBCL20(4),TBCL20(17)
      1,1,1,10,3,10,3)
       COEL2=TBLU2(A,CTTOT,TBCL40(7),TBCL40(4),TBCL40(17)
      1,1,1,10,3,10,3)
  150  CL=(COEL2-COEL1)*T+COEL1
       GO TO (10,200,30,40) I
C  CALCULATION OF THE GROUND EFFECTS UPDATE FOR
C  THE LIFT COEFFICIENT.
   10  CLWBH=CL
       CLH=TBLU2(TAILA,ELEOL,TBCLH(7),TBCLH(4),TBCLH(15)
      1,1,1,8,3,8,3)
C*********CLFWB=IS THE WING-BODY LIFT COEFFICIENT INCLUDING
C          FLAP EFFECTS, OUT OF GROUND EFFECT.
       CLFWB=CLWBH-CLH*SRATIO
       X=TBLU1(H75,TBX(4),TBX(15),1,11)
       IF(X.LT.0.0) X=0.0
       ALUP=AL VA+.5
       ALDN=AL VA-.5
       EUP=ALUP*(.24-.0015*FLAPS)
       EDN=ALDN*(.24-.0015*FLAPS)
       AHUP=ALUP-EUP
       AHDN=ALDN-EDN
       A=ALUP
       GO TO 5
  200  CLUP=CL
       A=ALDN
       GO TO 5
   30  CLDN=CL
       CLHUP=TBLU2(AHUP,ELEOL,TBCLH(7),TBCLH(4),TBCLH(15)
      1,1,1,8,3,8,3)
       CLHDN=TBLU2(AHDN,ELEOL,TBCLH(7),TBCLH(4),TBCLH(15)
      1,1,1,8,3,8,3)
C*********CLAWB=IS THE WING-BODY LIFT CURVE SLOPE,PER DEGREE
C          OUT OF GROUND EFFECT. IT IS EVALUATED BY CALCULATING
C          CLFWB FOR AL VA+.5 AND AL VA-.5 DEGREES. CLAWB IS
C          ASSIGNED THE AVERAGE SLOPE CALCULATED FOR THIS
C          ONE DEGREE CHANGE IN ALPHA.
       CLAWB=CLUP-CLDN-SRATIO*(CLHUP-CLHDN)
       CKCL=CLWBH*9.1196
       OLL=TBLU2(H25,CKCL,TBLLO(7),TBLLO(4),TBLLO(19)
      1,1,1,-12,3,12,3)
       R=SQRT(1.+H75**2.)-H75
       DELDEL=TBLU1(H25,TBDEL(4),TBDEL(9),1,5)
       IF(DELDEL.GT.0.0) DELDEL=0.0
       CK11=9.12/ASPECT+7.16*CRB
       CK12=ASPECT*CRB/2.
       CK13=(FLAPS/50.)**2.
```

5

Figure 135 XC-8A Aircraft Simulation Model File (Continued)

```
C*********DELAL=THE CHANGE IN ALPHA AT THE PRESENT
C         ALTITUDE.
      DELAL=-CK11*CLFWB*X-(CK12/CLAWB)*OLL*CLFWB*R-CK13*DELDEL/CLAWB
      IF(DELAL.GT.0.0) DELAL=0.0
      ALGR=AL VA-DELAL
      A=ALGR
GO TO 5
  40  CL1=CL
      DELCLGR=CLWBH-CL1
      ZO OL=ZO OL+DELCLGR
C  CALCULATION OF THE GROUND EFFECT UPDATE FOR
C  THE MOMENT M COEFFICIENT.
      FLAPNO=TBLU2(AL VA,CTTOT,TBCLO(7),TBCLO(4),TBCLO(17)
     1,1,1,10,3,10,3)
C*********CLWB=THE WING-BODY LIFT COEFFICIENT, FLAPS RETRACTED,
C         OUT OF GROUND EFFECT.
      CLWB=FLAPNO-SRATIO*CLH
C*********DELCLF=THE CHANGE IN LIFT COEFFICIENT DUE TO FLAPS,
C         OUT OF GROUND EFFECT.
      DELCLF=CLWBH-FLAPNO
C*********BEFF=THE EFFECTIVE WING SPAN.
      BEFF=(CLWB+DELCLF)/((CLWB/74.4)+(DELCLF/58.8))
      BEFF2=BEFF**2.
      SN1=BEFF2+4.*(HTAIL-H25CR)**2.                                     5
      SN2=BEFF2+4.*(HTAIL+H25CR)**2.
      DELE=EPSILON*SN1/SN2
      HAL=TAILA+DELE
      CLHGR=TBLU2(HAL,ELEOL,TBCLH(7),TBCLH(4),TBCLH(15)
     1,1,1,8,3,8,3)
      DELCLHG=CLHGR-CLH
C*********DELCMHG=THE CHANGE IN PITCHING MOMENT COEFFICIENT
C         PRODUCED BY THE HORIZONTAL STABILIZER. A .4MAC
C         C.G. LOCATION WAS ASSUMED.
      DELCMHG=-DELCLHG*SRATIO*4.304
      DCLWBG=DELCLGR+DELCLHG*SRATIO
C*********DCMUBG=THE CHANGE IN PITCHING MOMENT FOR THE
C         WING. A .4C C.G. LOCATION WAS ASSUMED.
      DCMWBG=-.15*DCLWBG
      DELCMG=DCMWBG+DELCMHG
      MO OL=MO OL+DELCMG
C CALCULATION OF THE GROUND EFFECTS UPDATE FOR
C  THE DRAG COEFFICIENT.
      SIGMA=EXP(-2.48*H75**.768)
      DELCDLG=SIGMA*(CLFWB**2.)/(ASPECT*3.14159)
      XO OL=XO OL+DELCDLG
 777  CONTINUE
C*********************************************************************
C*********************************************************************
LOCATION=34      OL      INPUTS=VA
```

Figure 135 XC-8A Aircraft Simulation Model File (Continued)

```
LOCATION= 1     DL      INPUTS=VA,OL
LOCATION=40     SG      INPUTS=OL,DL
END OF MODEL
PRINT
```

Figure 135 XC-8A Aircraft Simulation Model File (Concluded)

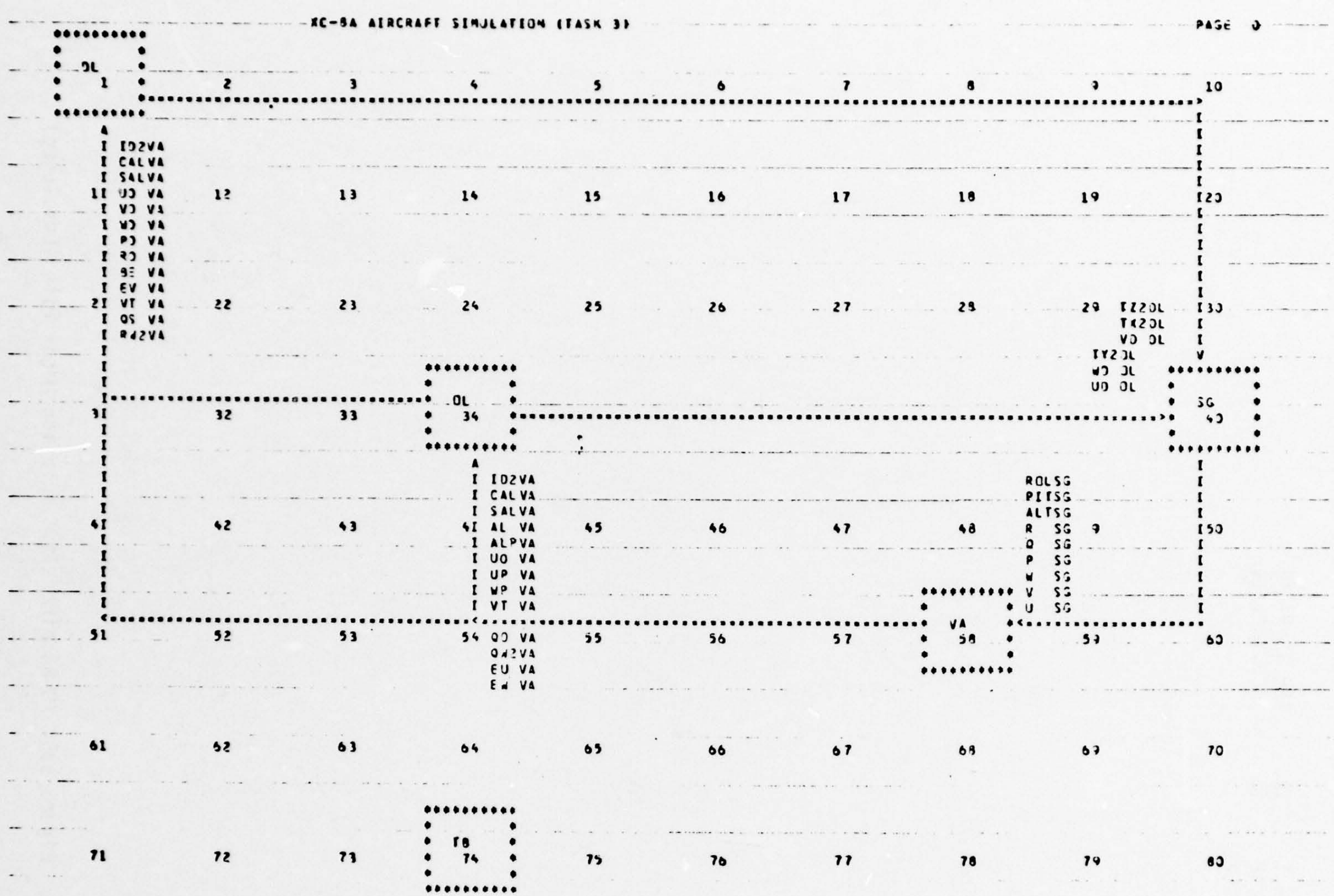


Figure 136 Easy XC-8A Aircraft Simulation Model Schematic

The model file is divided into five groups of Fortran coding. These groups are labeled 1 through 5 in Figure 135. The function of each of these groups are described below. User input parameters required by these Fortran statements are specified and defined with comment cards at the beginning of each group of statements.

1. These Fortran statements update the sine and cosine of the angle of attack with the current angle of attack. The internal coding in component VA evaluates these sine and cosine terms with the trim angle of attack (ALSVA) which is a constant. This simplification is valid for small perturbations in angle of attack about ALSVA. Since the variations in the angle of attack are expected to be significant for the XC-8A simulations the small perturbation assumption is then invalid. Thus to assure proper resolution of the aerodynamic forces from the stability axis into the body axis the sine and cosine terms must be updated with the current angle of attack. Variables PO VA and RO VA are also updated to reflect the update in the sine and cosine values.

   Since these Fortran statements modify the output from component VA, they must be positioned after VA but before OL, DL or any other component which uses these terms.

2. These Fortran statements limit control surface deflections and engine thrust to the maximum and minimum values attainable in the XC-8A aircraft. The sign convention for the control surface deflections are also specified by comment cards in this section.

3. These Fortran statements resolve the engine thrust into body axis forces and moments. The user can input forward, reverse and differential thrust values.

The side force (FY1DL), the rolling moment (TX1DL) and yawing moment (TZ1DL) incorporate the aerodynamic effects described in Reference 7 and implemented in the Air Force XC-8A Flight Simulator (see next item).

4. These Fortran statements determine the aerodynamic coefficients and derivatives required as input parameters by components OL and DL. This is the same mathematical model derived in Reference 7, for the Air Force XC-8A Flight Simulator. The aerodynamic effects produced by the trunk, ASP-10s, and wing skids have been included in this section.

   This coding along with the coding described in item 1 transforms the constant coefficient EASY aerodynamic model into a nonlinear variable coefficient model.

5. This final grouping of Fortran statements determines the aerodynamic ground effects for the XC-8A model. The procedure used to evaluate the ground effects for the model is defined in Reference 8, Method 1.

8.2.2 Analysis File

The analysis file for the XC-8A aircraft simulation is listed in Figure 137. The nine tables input in this file are described below:

TBCL0, TBCL20, TBCL40 are tabularized coefficients of lift, for 0, 20 and 40 degree flaps respectively versus the angle of attack (primary) and total coefficient of thrust (secondary) as shown in Figures 138, 139 and 140.

TBCLH are tabularized coefficients of lift for the horizontal stabilizer versus the horizontal stabilizer angle of attack (primary) and elevator deflection (secondary) are shown in Figure 141. Curves for 15, 0 and -25 degree elevator deflections were input in table TBCLH. A linear interpolation is performed for elevator deflections between these values.

```
TABLE=TBCL0,10,3
0.0,0.8,1.2
-4.0,-2.0,0.0,2.0,4.0,6.0,8.0,10.0,12.0,14.0
-.0154,.1523,.3378,.5153,.6988,.8734,1.0385
1.1809,1.3020,1.4271
-.0480,.1909,.4496,.7179,.9861,1.2423,1.4788
1.6915,1.8735,2.0650
-.0692,.2094,.4975,.7855,1.0790,1.3720,1.6264
1.8644,2.0840,2.2780
TABLE=TBCL20,10,3
0.0,0.8,1.2,
-4.0,-2.0,0.0,2.0,4.0,6.0,8.0,10.0,12.0,14.0
1.0055,1.1849,1.3435,1.4986,1.6568,1.8134
1.9626,2.0934,2.2089,2.3156
1.4464,1.7130,1.9513,2.1828,2.3979,2.6214
2.8429,3.0386,3.2056,3.3703
1.6034,1.8913,2.1564,2.3988,2.6470,2.8904
3.1287,3.3488,3.5450,3.7086
TABLE=TBCL40,10,3
0.0,0.8,1.2
-4.0,-2.0,0.0,2.0,4.0,6.0,8.0,10.0,12.0,14.0
1.6639,1.8323,1.9766,2.1411,2.3094,2.4749
2.5999,2.7305,2.8406,2.9339
2.4030,2.6580,2.8879,3.1195,3.3492,3.5917
3.7955,3.9699,4.1346,4.2591
2.6590,2.9325,3.1828,3.4399,3.7065,3.9480,4.1946
 4.4070,4.5844,4.7244
TABLE=TBX,11
0.0,.1,.2,.3,.4,.5,.6,.7,.8,.9,1.0
1.0334,.7145,.5360,.4176,.3239,.2586,.2080
.1669,.1352,.1087,.0899
TABLE=TBCLH,8,3
-25.0,0.0,15.0
-18.0,-16.0,-14.0,8.0,10.0,12.0,14.0,16.0
-1.6755,-1.6289,-1.5699,-.3587,-.2409
-.1127,.00755,.1004
-1.0937,-1.0272,-.9288,.4601,.5818
.7110,.7566,.6414
-.6462,-.5369,-.4162,1.0091,1.1333,1.0773
.9537,.8311
TABLE=TBLL0,12,3
10.0,18.0,24.0
.2,.4,.6,.8,1.0,1.2,1.4,1.6,1.8,2.0,2.2,2.4
.2118,.06113,.00509,-.01720,-.02592,-.02780,-.02880,-.02938
-.02902,-.02856,-.02814,-.02824
-.05664,-.09657,-.1098,-.1063,-.09462,-.08550,-.07764,-.07006
-.06439,-.06119,-.05832,-.05740
-.1942,-.1980,-.1855,-.1648,-.1426,-.1250,-.1114
-.09891,-.08937,-.08117,-.07524,-.07178
```

Figure 137 XC-8A Aircraft Simulation Analysis File

```
TABLE=TBDEL,5
.2,.4,.6,.8,1.0
-.09898,-.06246,-.03709,-.01953,-.005234
TABLE,A2TTB,5
0.0,2.99,3.0,5.0,30.0
-.037,-.037,-10.0,-.037,-.037
TABLE,B2TTB,2
0.0,30.0
0.0,0.0
PARAMETER VALUES
THRRT=0.0,THRLT=0.0
FSENG=197.0,BLENG=183.0,WLENG=191.0
FSCG=358.2,BLCG=0.0,WLCG=160.0
FS75CR=406.81,WL75CR=206.06,FS25CR=336.79
WL25CR=206.06,FSTAIL=889.65,WLTAIL=403.84
ROOTC=11.67,ASPECT=9.75,SH=233.0
THETA=0.0,TRUNK=0.0
CGRAV=40.0,SPOIL=0.0,FLAPS=0.0
AILDL=0.0
ID1VA=3,VS VA=253.0,ALSVA=3.39
S  VA=945.0,IDGVA=6
MA1OL=1274.5,C  OL=10.29,ISWOL=3
B  DL=96.0
IXXSG=301300.,IYYSG=267000.
IZZSG=508600.,IXZSG=24190.
INITIAL CONDITIONS
U  SG=252.56,V  SG=0.0,W  SG=14.96
P  SG=0.0,Q  SG=0.0,R  SG=0.0
ROLSG=0.0,PITSG=3.39,YAWSG=0.0
ALTSG=5000.0,X  SG=0.0,Y  SG=0.0
ERROR CONTROLS
U  SG=.0001,V  SG=.0001,W  SG=.0001
P  SG=.0001,Q  SG=.0001,R  SG=.0001
ROLSG=.0001,PITSG=.0001,YAWSG=.0001
ALTSG=.0001,X  SG=.0001,Y  SG=.0001
PRINT CONTROL=4
ALL STATES
LINEAR ANALYSIS
XIC1-XIC
TINC=.5,TMAX=20.0,INT MODE=2
PRINTER PLOTS,PLOT ON
DISPLAY1
VT VA,VS,TIME
U  SG,VS,TIME
W  SG,VS,TIME
ALTSG,VS,TIME
ELEOL,VS,TIME
DISPLAY2
AL VA,VS,TIME
```

Figure 137 XC-8A Aircraft Simulation Analysis File (Continued)

```
GAMMA,VS,TIME
PITSG,VS,TIME
Q  SG,VS,TIME
ELEOL,VS,TIME
PLOT ID   PAUL R. PERKINS  MS 47-03
TITLE    XC-8A LONGITUDINAL STABILITY EVALUATION (TASK 3)
SIMULATE
TABLE,A2TTB,2
0.0,30.0
-.037,-.037
TABLE,B2TTB,5
0.0,2.99,3.0,5.0,30.0
0.0,0.0,10.0,0.0,0.0
XIC-XIC1
TITLE   XC-8A LATERAL STABILITY EVALUATION (TASK 3)
DISPLAY1
V  SG,VS,TIME
BE VA,VS,TIME
R  SG,VS,TIME
RUDDL,VS,TIME
SIMULATE
```

Figure 137 XC-8A Aircraft Simulation Analysis File (Concluded)

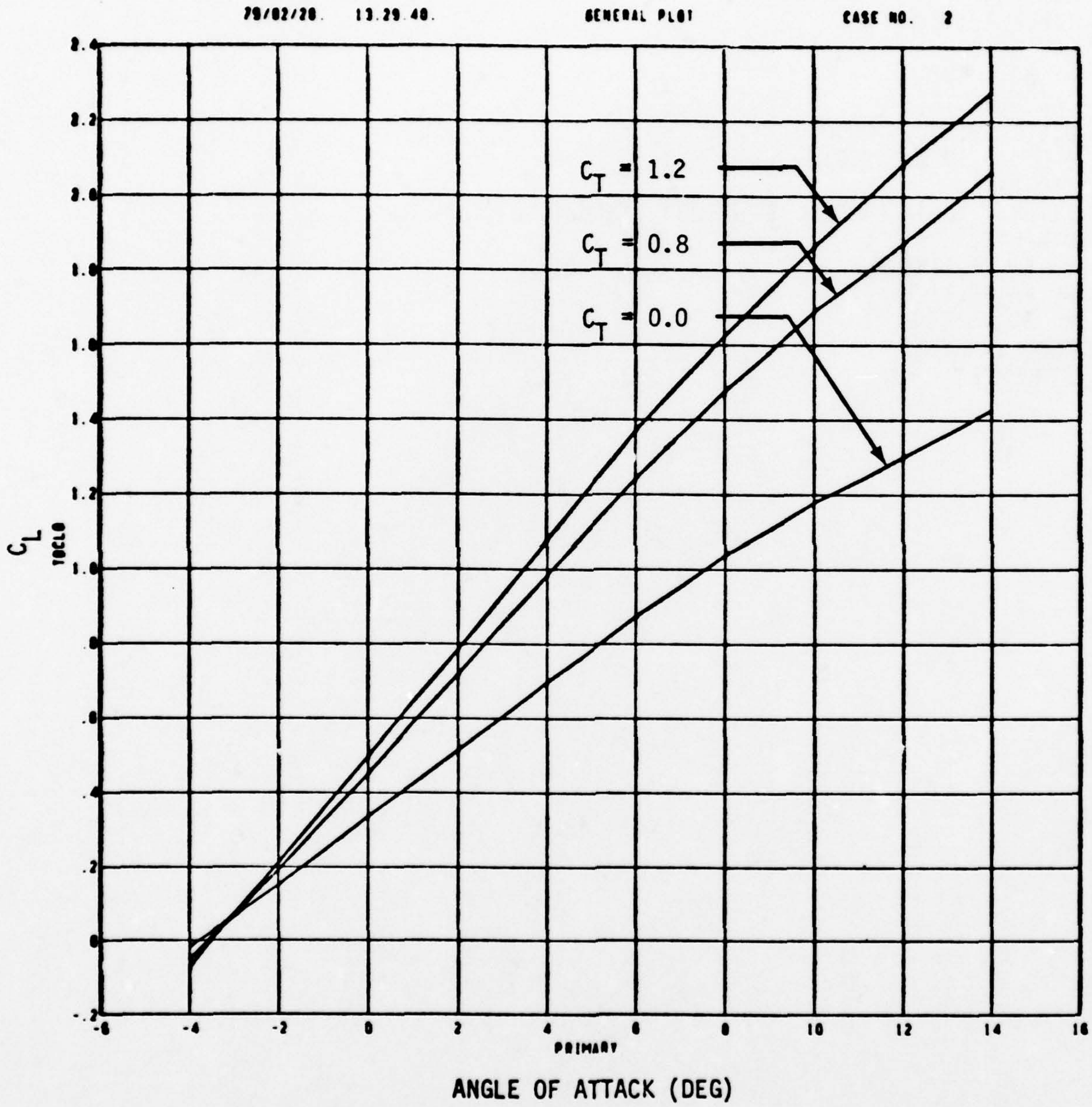


Figure 138 Table TBCLO, C_L for 0 degree Flaps

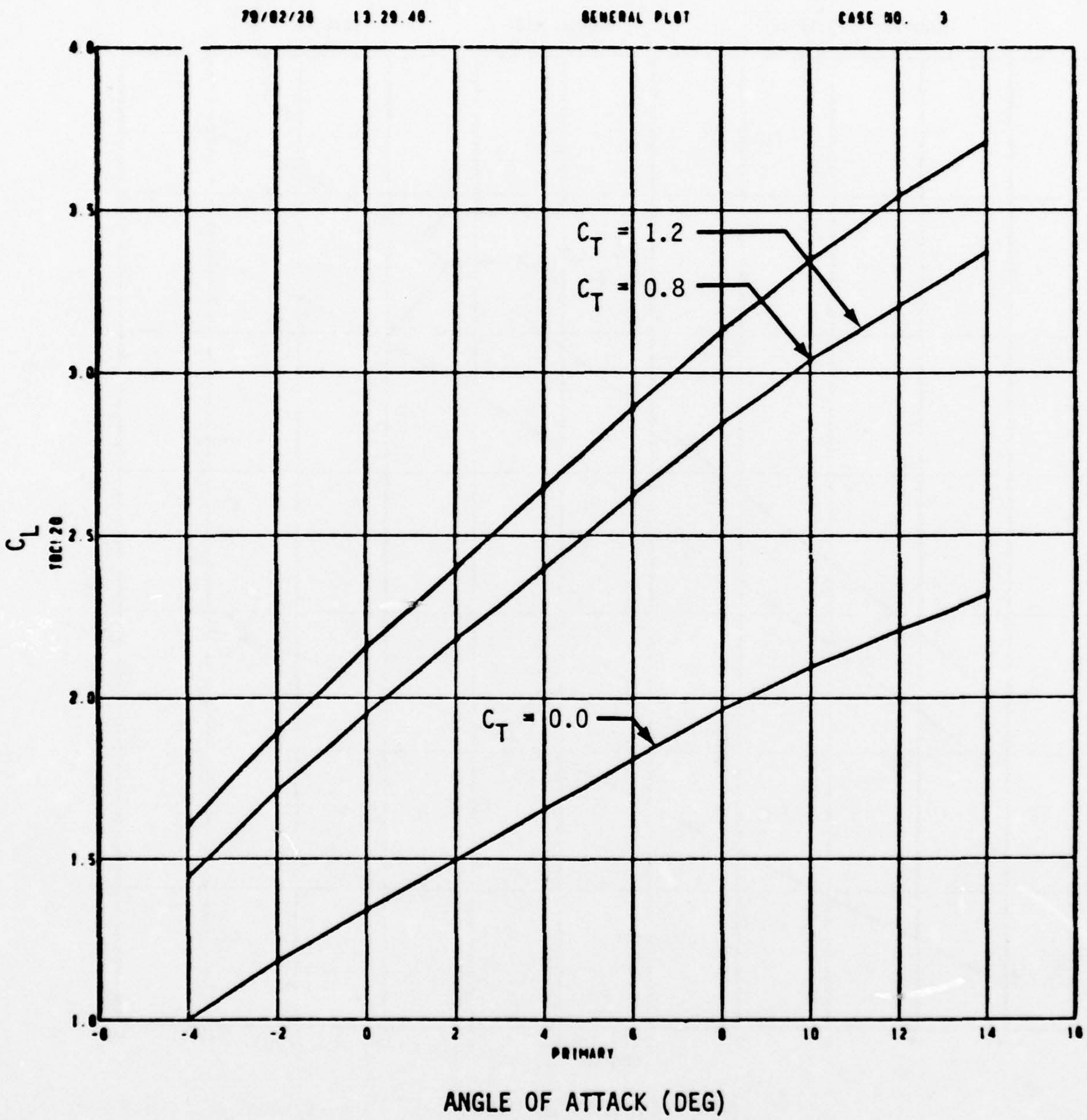


Figure 139 Table TBCL20, C_L for 20 Degree Flaps

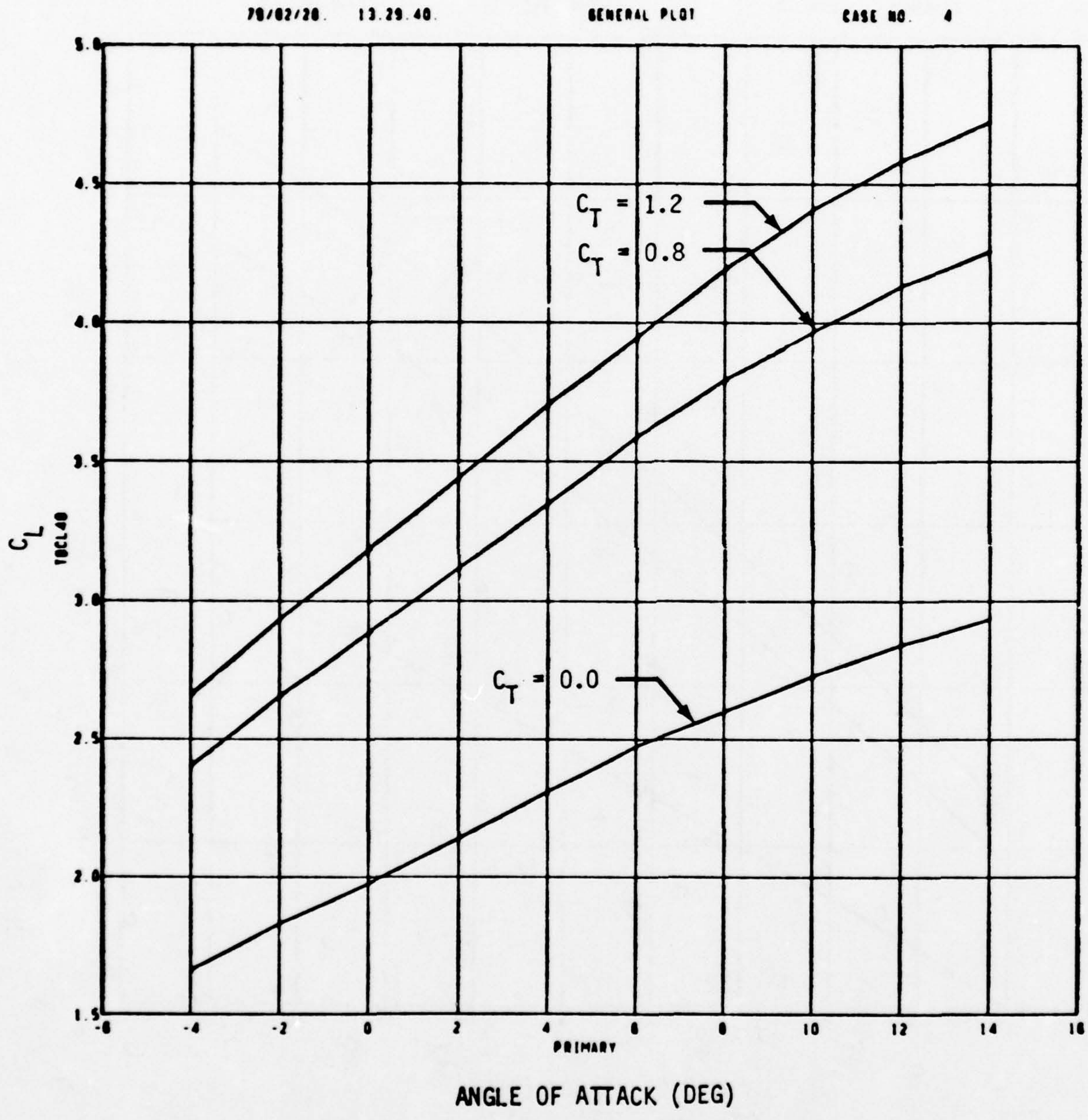


Figure 140 Table TBCL40, C_L for 40 Degree Flaps

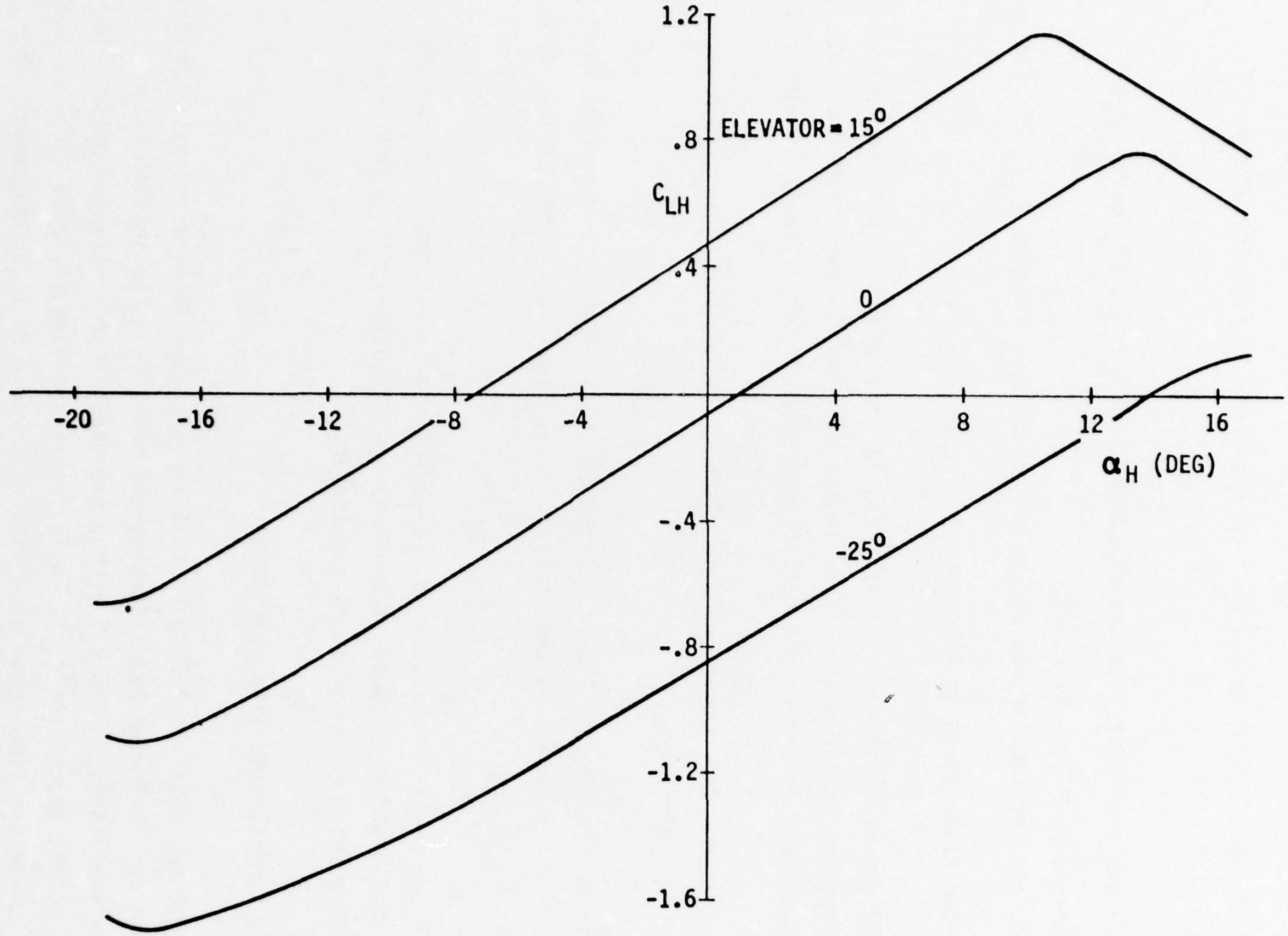


Figure 141 Table TBCLH, Lift Coefficient for the Horizontal Stabilizer

TBX are tabularized values of X versus h/(b/2) given in Reference 8, Figure 4.7.1-14.

TBLLO are tabularized values of L/LO-1 versus $h_{.25C_r}/C_r$ given in Reference 8, Figure 4.7.1-15.

TBDEL are the tabularized values of $\Delta(\Delta C_L)_{FLAP}$ versus $h_{.25C_r}/C_r$ for slotted flaps given in Reference 8, Figure 4.7.1-17.

A2TTB and B2TTB are respectively tabularized values of elevator or rudder deflections versus time. These tables are used to input the saw tooth shaped elevator and rudder deflections used in the aircraft simulation.

The tabularized values input by tables TBCLO, TBCL20, TBCL40 and TBCLH were obtained from Reference 9.

For convenient reference, Table 10 lists the basic XC-8A aircraft data which were input as parameters in the analysis file. This data were compiled from References 9, 10 and 11.

Since all the models states, rates, variables and parameters are known the model initialization is complete and a steady state analysis is not required prior to executing a time history simulation.

8.3 Longitudinal Stability Evaluation

To validate the EASY XC-8A lingitudinal aerodynamics, a longitudinal stability evaluation was performed by perturbing the model from the initial flight configuration listed in Table 9 with a saw tooth shaped elevator pulse. The elevator pulse, input from table A2TTB, decreased linearly from -.037 to -10.0 degrees during the simulation time interval from 2.99 to 3.0 seconds. Then from 3.0 to 5.0 seconds the elevator increased linearly back to -.037 degrees. All the states were active during the longitudinal stability evaluation.

TABLE 10

XC-8A AIRCRAFT DATA

MEAN CHORD LENGTH (C)	10.29 FT
ROOT CHORD LENGTH	11.67 FT
WING SPAN (b)	96.0 FT
WING AREA (S)	945.0 FT^2
ASPECT RATIO (A)	9.75
HORIZONTAL STABILIZER AREA (SH)	233 FT^2
THRUST LOCATION FOR RIGHT ENGINE AT PROPELLER HUB:	
FSENG	197.0 IN
BLENG	183.0 IN
WLENG	191.0 IN
ANGLE OF THRUST W.R.T WATERLINE	0.0 DEG
WING SKID LOCATION (RIGHT SKID):	
FSSKID	340.0 IN
BLSKID	432.5 IN
WLSKID	90.0 IN
LOCATION OF .25 ROOT CHORD:	
FS25CR	336.79 IN
BL25CR	0.0 IN
WL25CR	206.06 IN
LOCATION OF .75 ROOT CHORD:	
FS75CR	406.81 IN
BL75CR	0.0 IN
WL75CR	206.06 IN
LOCATION OF .25 CHORD FOR HORIZONTAL STABILIZER:	
FSTAIL	889.65 IN
WLTAIL	403.84 IN

Time histories generated by the EASY XC-8A simulation and the Air Force XC-8A Flight Simulator for pitch, angular velocity Q, angle of attack, air speed and altitude are shown respectively in Figures 142a, b, c, d, and e. Comparison of these data demonstrate an excellent point by point correlation between the XC-8A Flight Simulator and the EASY XC-8A simulation, thereby validating the EASY longitudinal aerodynamics.

8.4 Lateral Stability Evaluation

To validate the EASY XC-8A lateral aerodynamics, a lateral stability evaluation was performed by perturbing the model from the initial flight configuration listed in Table 9 with a saw tooth shaped rudder pulse. The rudder pulse, input from table B2TTB, decreased linearly from 0.0 to -10.0 degrees during the simulation time interval from 2.99 to 3.0 seconds. Then from 3.0 to 5.0 seconds the rudder increased linearly back to 0.0 degrees. All the states were active during the lateral stability evaluation.

Time histories generated by the EASY simulation and the Air Force XC-8A Flight Simulator for v velocity, beta and yaw rate are shown respectively in Figures 143a, b, and c. Comparison of these data demonstrate an excellent point by point correlation between the XC-8A Flight Simulator and the EASY XC-8A simulation, thereby validating the EASY lateral aerodynamics.

8.5 Conclusions

Based on the excellent correlation of the EASY XC-8A aerodynamic model with the Air Force XC-8A Flight Simulator during the longitudinal and lateral stability evaluations the EASY models used to simulate the XC-8A aircraft in flight maneuvers are considered to be valid.

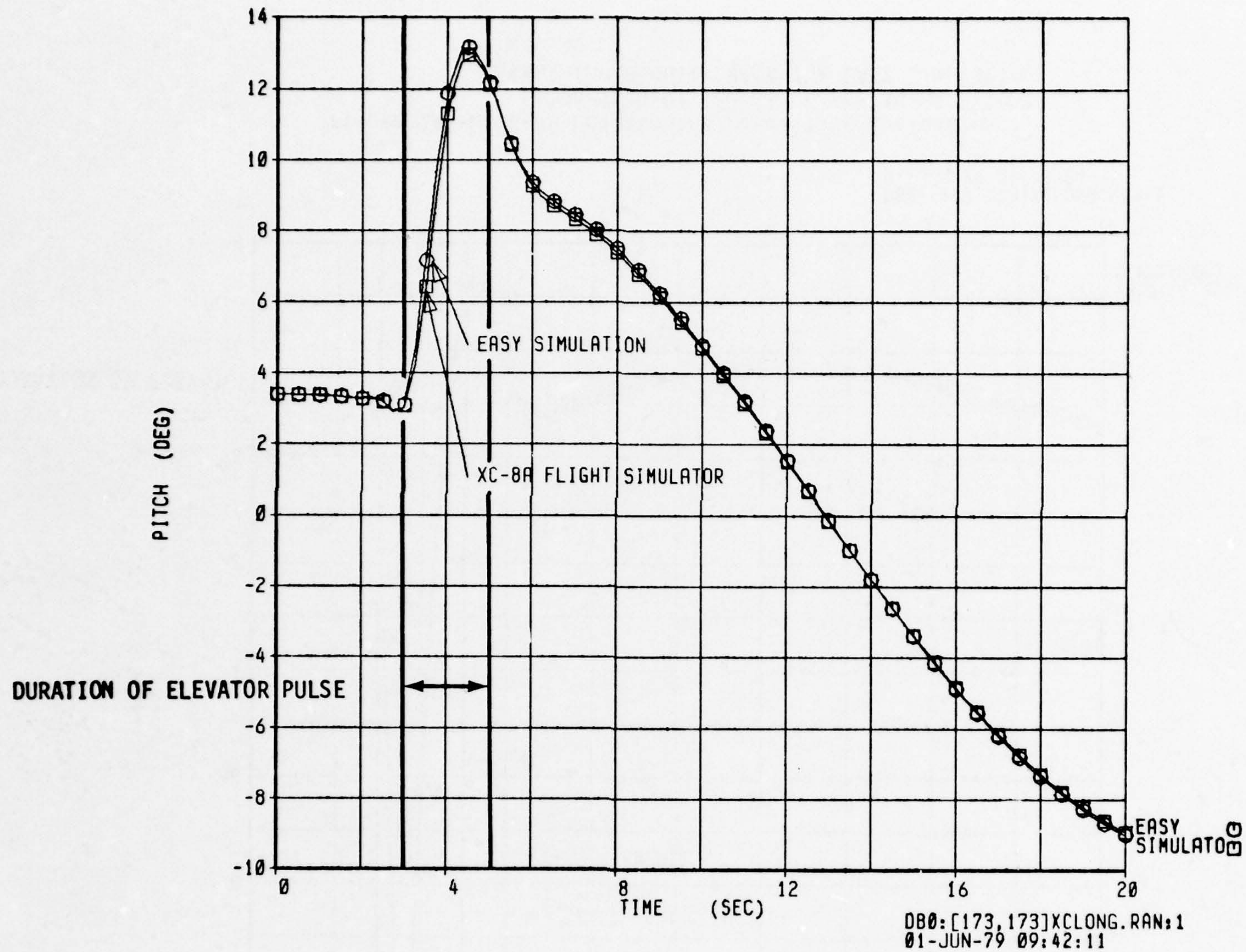


Figure 142-a XC-8A Longitudinal Stability Evaluation
A Comparison Of The Air Force XC-8A Flight Simulator Results With The EASY Simulation Results

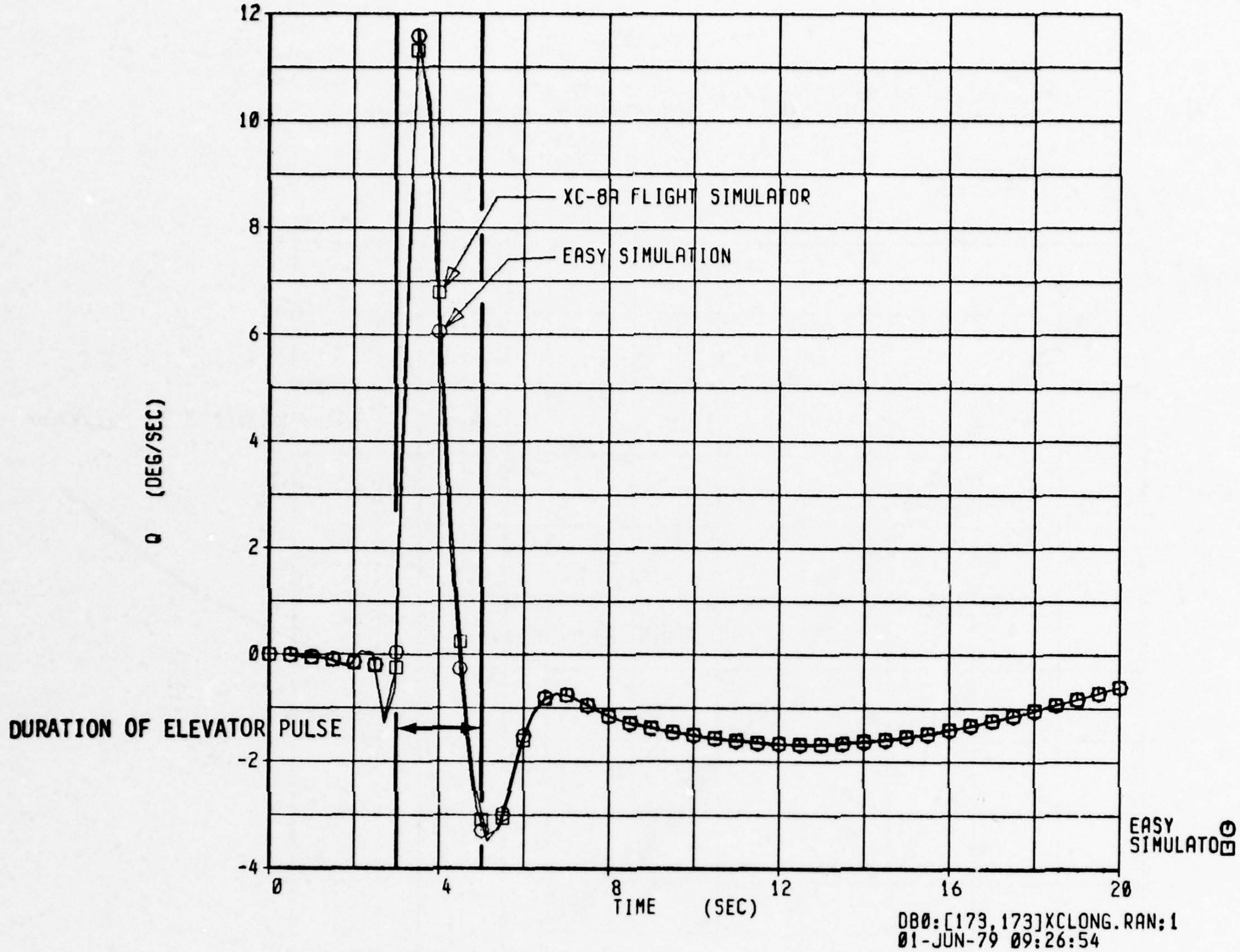


Figure 142-b XC-8A Longitudinal Stability Evaluation
A Comparison Of The Air Force XC-8A Flight Simulator Results With The EASY Simulation Results

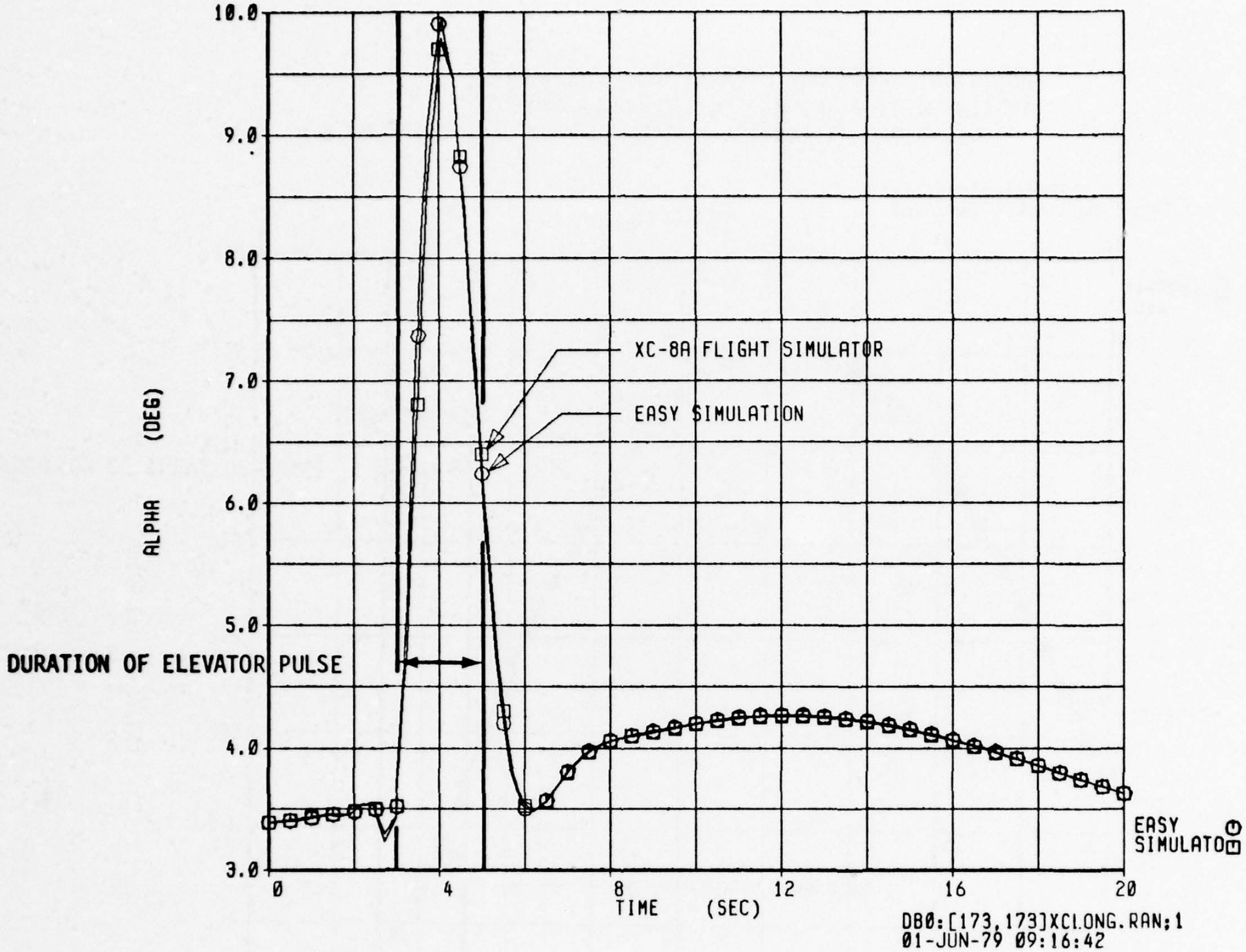


Figure 142-c XC-8A Longitudinal Stability Evaluation
A Comparison Of The Air Force XC-8A Flight Simulator Results With The EASY Simulation Results

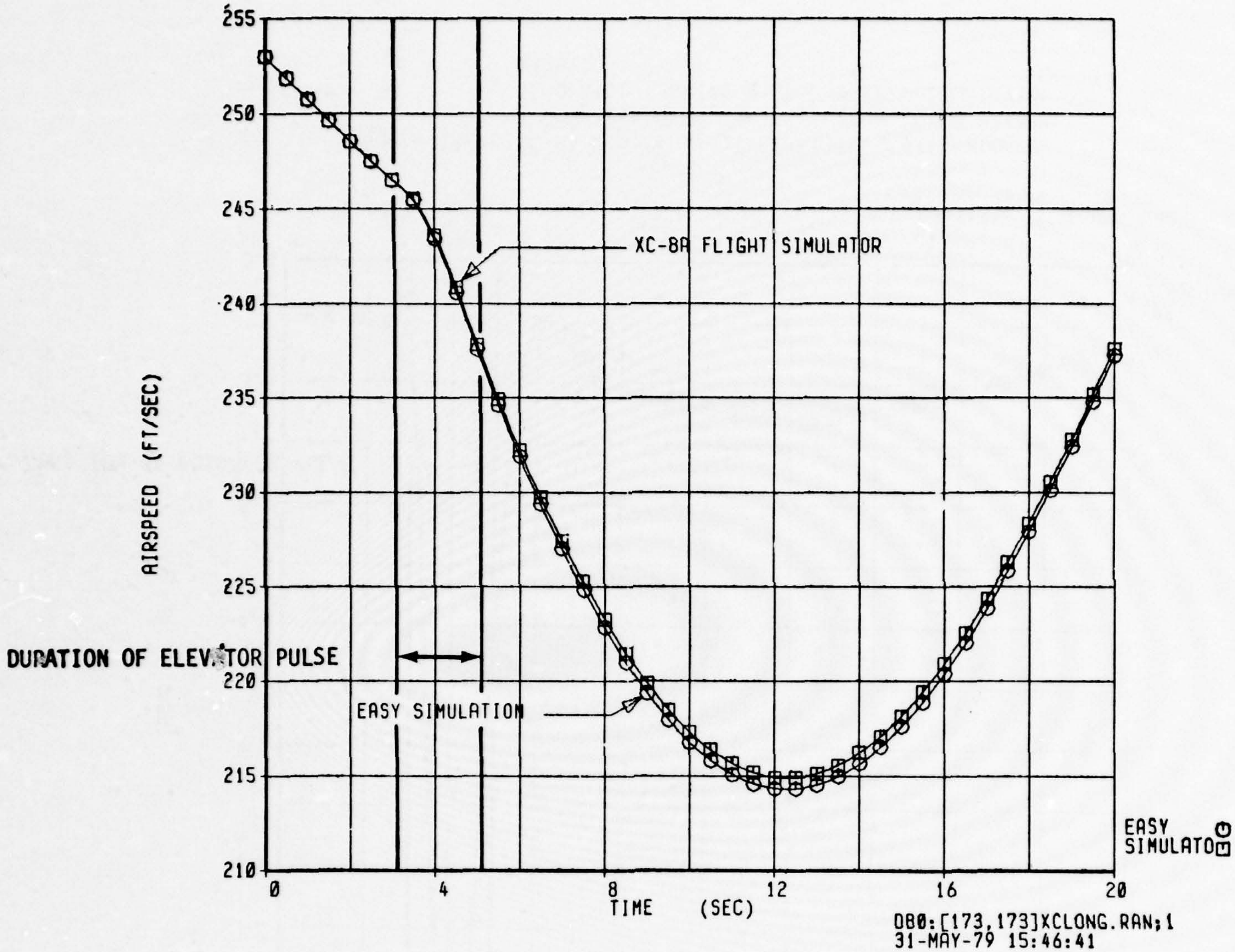


Figure 142-d XC-8A Longitudinal Stability Evaluation
A Comparison Of The Air Force XC-8A Flight Simulator Results With The EASY Simulation Results

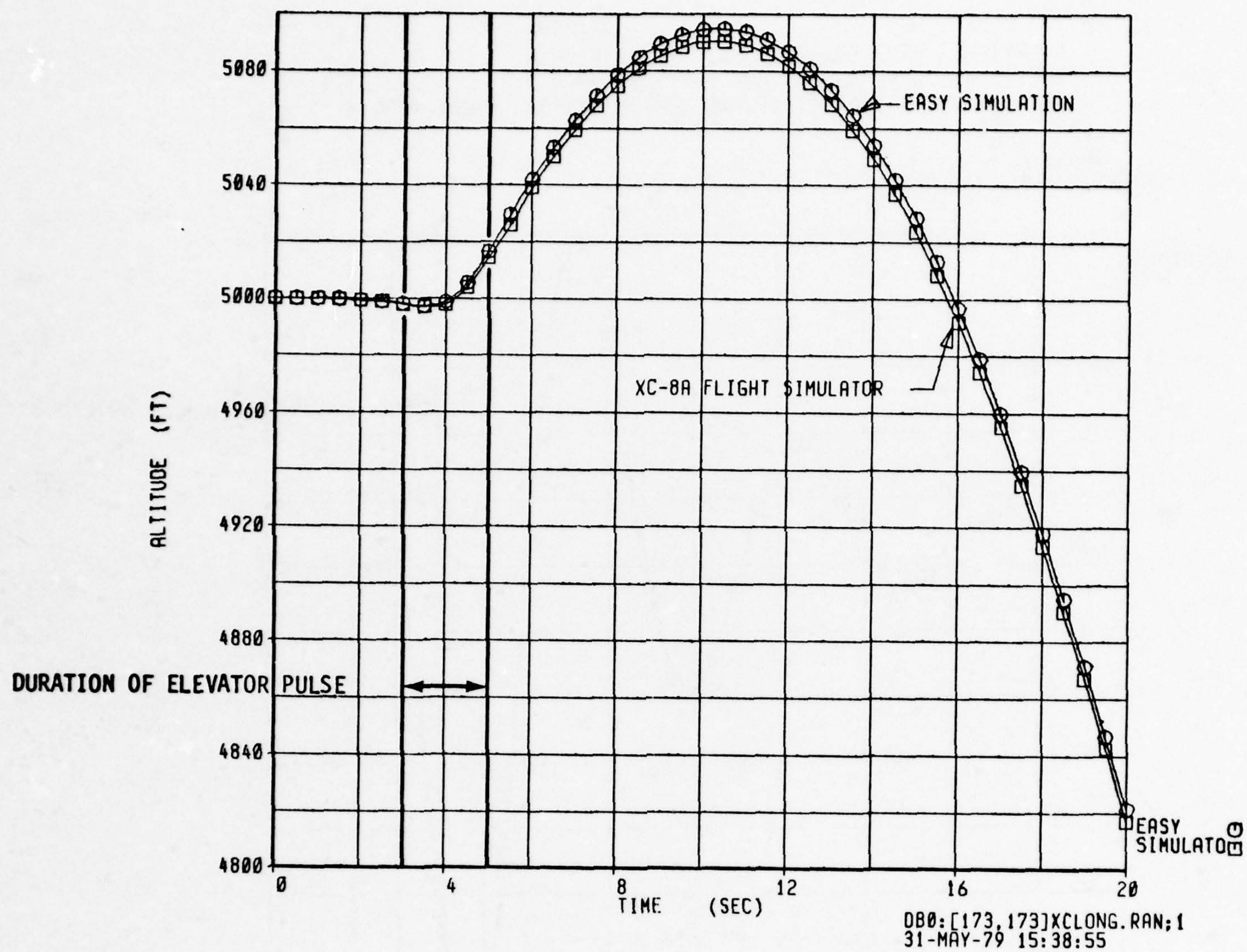


Figure 142-e XC-8A Longitudinal Stability Evaluation
A Comparison Of The Air Force XC-8A Flight
Simulator Results With The EASY Simulation
Results

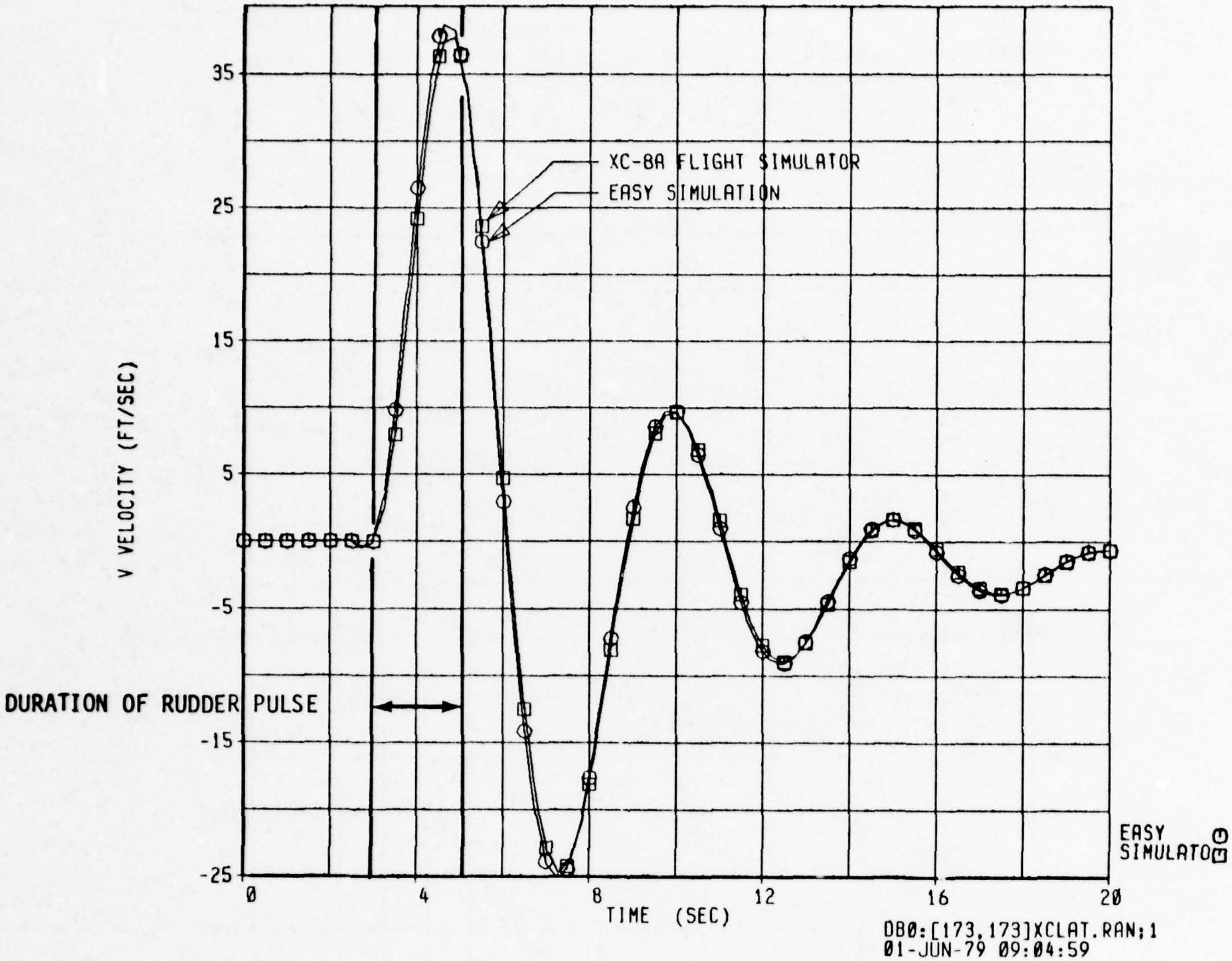


Figure 143-a XC-8A Lateral Stability Evaluation
A Comparison Of The Air Force XC-8A Flight Simulator Results With The EASY Simulation Results

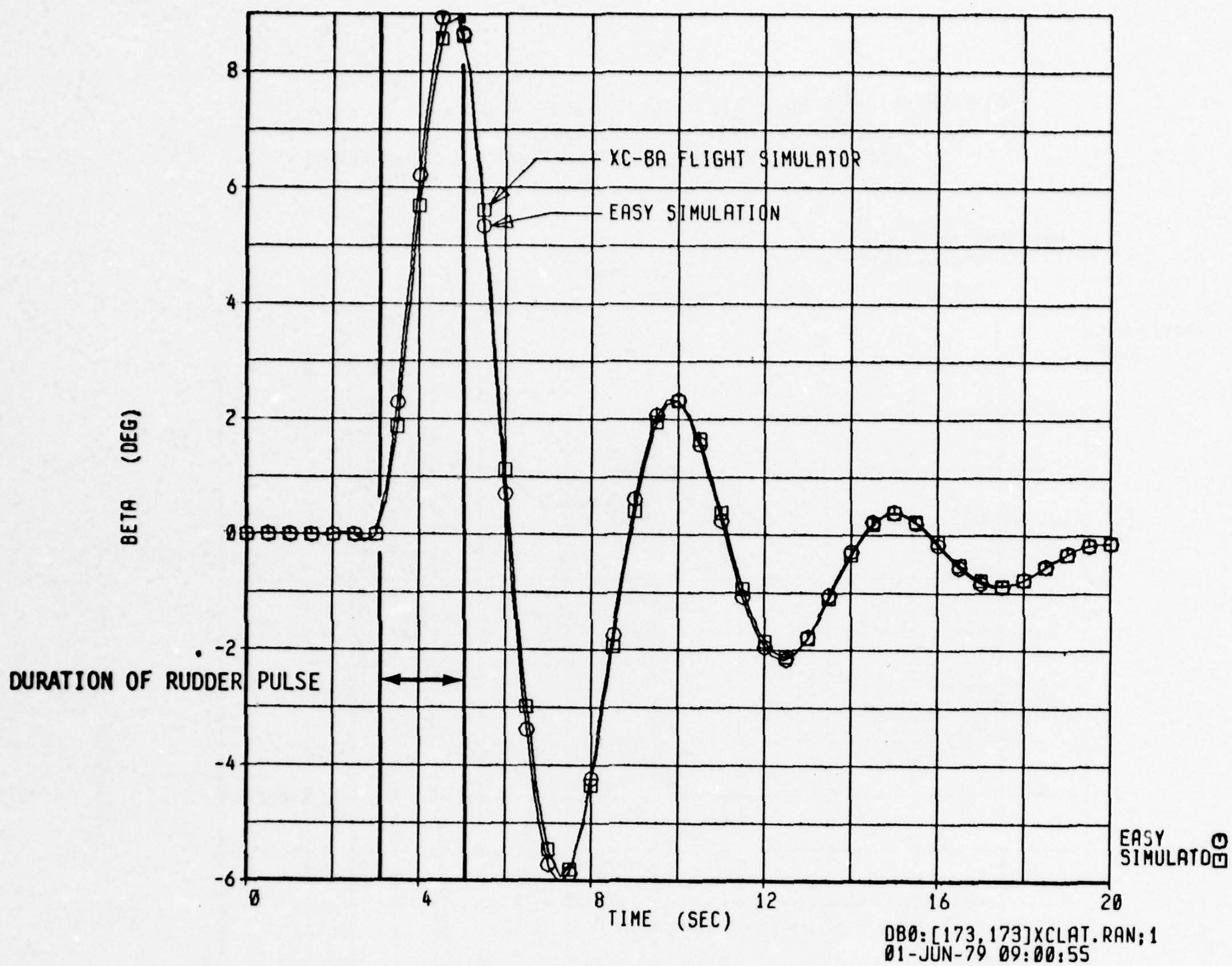


Figure 143-b XC-8A Lateral Stability Evaluation
A Comparison Of The Air Force XC-8A Flight Simulator Results With The EASY Simulation Results.

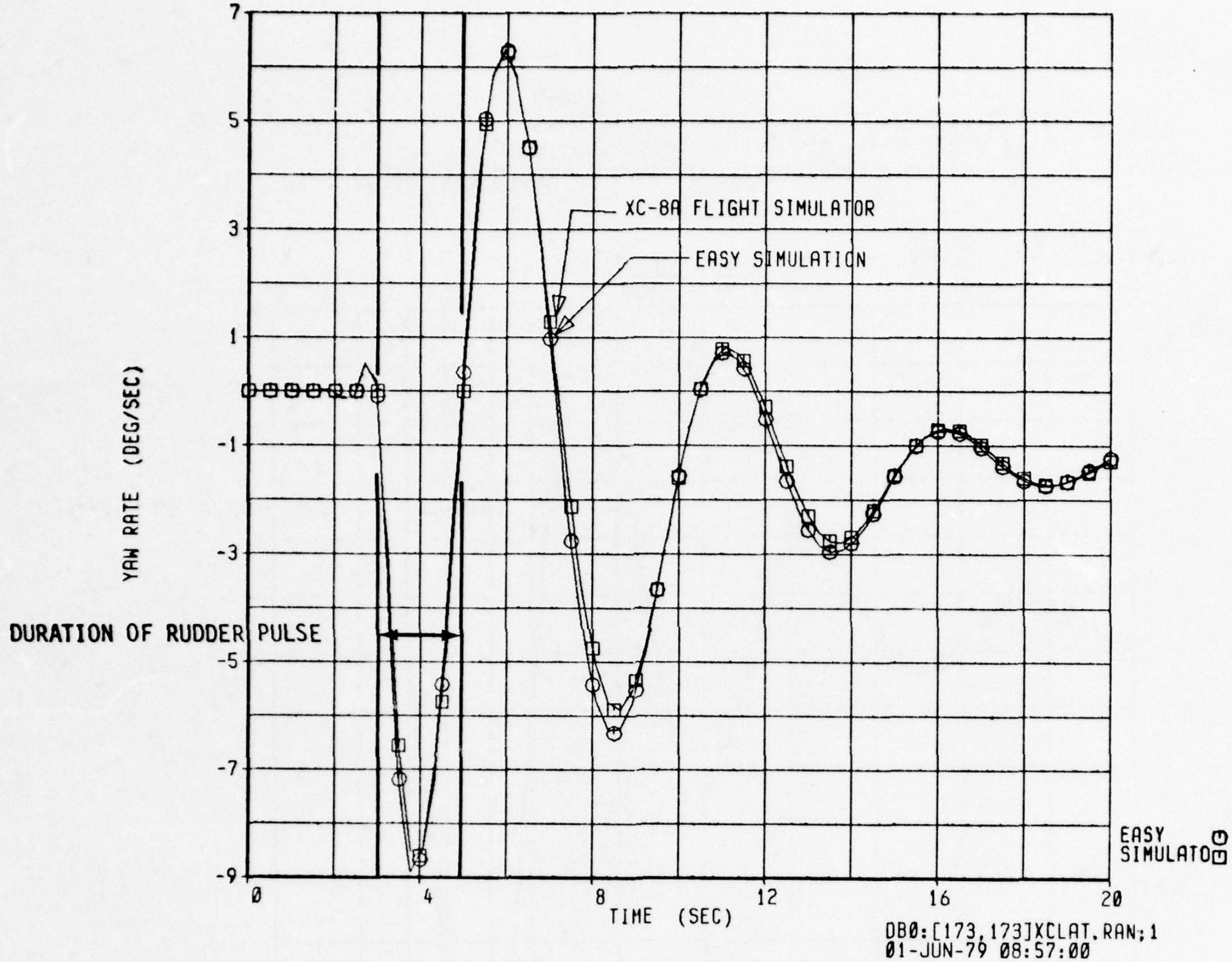


Figure 143-c XC-8A Lateral Stability Evaluation
A Comparison Of The Air Force XC-8A Flight Simulator Results With The EASY Simulation Results

SECTION IX

XC-8A LANDING SIMULATION

9.1 Objectives

The objective of the landing simulation was to create an EASY computer model of the XC-8A ACLS aircraft incorporating the 6-DOF rigid body aerodynamic model developed for the XC-8A aircraft simulation, an elastic ACS trunk, two ASP-10 fans and two wing skids into the model. A typical XC-8A ACLS landing was simulated using the EASY model and that simulation was compared with the actual flight test data in an effort to validate the model.

9.2 Technical Approach

The XC-8A Test Flight #051276-1, landing #1 with a 8.0 ft/sec sink rate and 20 degree flaps was selected for simulation. A complete listing of the initial conditions (corresponding to time=092527.9 in the flight test data) input into the EASY XC-8A ACLS model for this landing are listed in Table 11.

9.2.1 Model File

Thirteen standard EASY components along with several pages of Fortran coding were assembled to form the EASY XC-8A ACLS landing simulation model. The EASY XC-8A ACLS model file and computer generated model schematic are given in Figures 144 and 145. The XC-8A aerodynamic model contained in Figure 144 is the same model implemented for the aircraft simulation. Only those items which were added to the aerodynamic model to form the landing model will be discussed here. The reader who is unfamiliar with the aerodynamics should refer to section 8.2.1 for background information. The implementation of the air cushion system with the aerodynamic model required the addition of

TABLE 11

XC-8A LANDING CONFIGURATION
(TIME = 27.9 SEC)

FLAPS	20 DEG
TRUE AIRSPEED	80.4 KNOTS
ALTITUDE (c.g.)	10 FT
GROSS WEIGHT	36000 LBS
CENTER OF GRAVITY (c.g.)	28% MAC
FSCG	343.41 IN
WLCG	160.0 IN
MOMENTS OF INERTIA:	
IXX	183700 SLUGS-FT^2
IYY	228200 SLUGS-FT^2
IZZ	365100 SLUGS-FT^2
IXZ	29220 SLUGS-FT^2
THRUST:	
THRRT	1487.2 LBS
THRLT	1487.2 LBS
ATTITUDE:	
ANGLE OF ATTACK (α)	6.21 DEG
SIDESLIP ANGLE (β)	0.0 DEG
FLIGHT PATH ANGLE (γ)	-3.31 DEG
ROLL (ϕ)	0.0 DEG
PITCH (θ)	2.9 DEG
YAW (ψ)	0.0 DEG
LONGITUDINAL CONTROL SURFACE DEFLECTIONS:	
ELEVATOR	-10.4 DEG
SPOILER	0.0 DEG

TABLE 11 (CONTINUED)

XC-8A LANDING CONFIGURATION

(TIME = 27.9 SEC)

LATERAL CONTROL SURFACE DEFLECTIONS:

RUDDER..................................	0.0 DEG
AILERON.................................	+.02 DEG

STATE RATES WHICH HAVE NONZERO VALUES:

PITCH...................................	2.0 DEG/SEC
$\dot{Q}$.......................................	2.3 DEG/SEC^2
ALTITUDE (SINK RATE)...................	-8.0 FT/SEC

AIR CUSHION VALUES:

TRUNK PRESSURE..........................	279.8 PSFG
CUSHION PRESSURE........................	8.2 PSFG
FAN RPM.................................	6186 RPM
EFFECTIVE AREA FOR TRUNK TO CUSHION TRIM VALVE..............................	329.86 IN^2

```
MODEL DESCRIPTION    XC-8A LANDING SIMULATION (TASK 5)
FORTRAN STATEMENTS
      COMMON /BMADTS/IERR
      REAL L1,M1,N1
      DATA IERR/0/
ADD PARAMETERS=THETA
FSENG,BLENG,WLENG,FSCG,BLCG,WLCG
FSSKID,BLSKID,WLSKID,SPLOW,SPHIGH,DEFMX,SKIDMU,SKID
TRUNK,CGRAV,FLAPS
ROOTC,ASPECT,SH,FS25CR,WL25CR
FS75CR,WL75CR,FSTAIL,WLTAIL
ADD VARIABLES=CTRT,CTLT,CTTOT,FFBETA
SPOIL,THRRT,THRLT
PCUSH,PTRUNK
COMP,ZFORCE,XFORCE
DELCLGR,DELCDLG,DELCMG
ACELCG,VKNOTS
ADD TABLES
TBCL0,46,TBCL20,46,TBCL40,46
TBCLH,38,TBX,-25,TBLL0,54,TBDEL,-13
ELEV=-15,SPOLER,-11
LOCATION=58     VA     INPUTS=SG
FORTRAN STATEMENTS
      RPD=.01745329
      FINIT1=8.0-VT VA*SIN((AL VA-PITSG)*RPD)
      FINIT3=V  SG
      FINIT4=P  SG
      FINIT5=16.627-PT TS
LOCATION=101     IT1
LOCATION=105     IT3
LOCATION=107     IT4
LOCATION=109     IT5
FORTRAN STATEMENTS
      THRRT=FO IT1
      THRLT=THRRT
      ELEOL=FO IT2
      RUDDL=FO IT3
      AILDL=FO IT4
      CAVTS=FO IT5
```

PSEUDO-INTEGRAL
CONTROLLERS FOR
MODEL INITIALIZATION

```
      IF(TIME.LE.27.9) GO TO 555
      ELEOL=TBLU1(TIME,ELEV(4),ELEV(10),1,-6)
 555  CONTINUE
      SPOIL=TBLU1(TIME,SPOLER(4),SPOLER(8),1,-4)
      SPBTS=SPOIL
```

ELEVATOR AND
SPOILER DEFLECTIONS
DURING SIMULATION

```
C*********VKNOTS=THE TRUE AIR SPEED. (KNOTS)
      VKNOTS=VT VA/1.6889
C*****************************************************************
C*****************************************************************
C  THESE FORTRAN STATEMENTS UPDATE THE COS(ALPHA) AND SIN(ALPHA)
```

Figure 144 EASY XC-8A ACLS Model File For The Landing Simulation

```
C  WITHOUT CHANGING THE VALUE OF ALSVA. THIS PROVIDES FOR CORRECT
C  RESOLUTION OF THE LIFT AND DRAG FORCES FROM THE STABILITY
C  AXES INTO THE BODY AXES. THESE FIVE FORTRAN STATEMENTS
C  MUST BE IMPLEMENTED WHENEVER THE AERODYNAMIC
C  COEFFICIENTS OR DERIVATIVES ARE INPUT AS FUNCTIONS
C  OF THE ANGLE OF ATTACK.
C  CAUTION- THESE FORTRAN STATEMENTS CANNOT BE USED WHEN THE
C  *******  AERODYNAMIC DERIVATIVES ARE REFERENCED TO THE BODY AXIS.
      RPD=.01745329
      SALVA=SIN(AL VA*RPD)
      CALVA=COS(AL VA*RPD)
      PO VA=P  SG*CALVA+R  SG*SALVA
      RO VA=R  SG*CALVA-P  SG*SALVA
C*********************************************************************
C*********************************************************************
C THE FOLLOWING IF STATEMENTS LIMIT THE ALLOWABLE DEFLECTIONS
C OF THE ELEVATOR,RUDDER,FLAPS AND AILERION WHEEL. THEY ALSO
C LIMIT THE ENGINE THRUST TO A SPECIFIED RANGE.
C*********RANGE OF ELEVATOR DEFLECTION= -25.0 TO +15.0 DEG.
C          (A POSITIVE ELEVATOR DEFLECTION IS DOWN)
      IF(ELEOL.LT.-25.0) ELEOL=-25.0
      IF(ELEOL.GT.15.0) ELEOL=15.0
C*********RANGE OF RUDDER DEFLECTION= -20.0 TO +20.0 DEG.
C          (A POSITIVE RUDDER DEFLECTION IS TO THE PILOTS LEFT)
      IF(RUDDL.LT.-20.0) RUDDL=-20.0
      IF(RUDDL.GT.20.0) RUDDL=20.0
C*********RANGE OF FLAP DEFLECTION= 0.0 TO +40.0 DEG.
      IF(FLAPS.LT.0.0) FLAPS=0.0
      IF(FLAPS.GT.40.0) FLAPS=40.0
C*********RANGE OF AILERION WHEEL ROTATION= -75.0 TO +75.0 DEG.
C          (A POSITIVE AILERION DEFLECTION TIPS THE RIGHT WING DOWN)
      IF(AILDL.LT.-75.0) AILDL=-75.0
      IF(AILDL.GT.75.0) AILDL=75.0
C*********THE MAXIMIUM THRUST PER ENGINE=+9000.0 LBS..
      IF(THRRT.GT.9000.0) THRRT=9000.0
      IF(THRLT.GT.9000.0) THRLT=9000.0
C*********THE MAXIMIUM REVERSE THRUST PER ENGINE=-9000.0
      IF(THRRT.LT.-9000.0) THRRT=-9000.0
      IF(THRLT.LT.-9000.0) THRLT=-9000.0
C*********************************************************************
C*********************************************************************
C  THESE FORTRAN STATEMENTS CALCULATE THE BODY AXIS FORCES
C  AND MOMENTS PRODUCED BY THE ENGINE THRUST. THE THRUST
C  LOCATION AND ANGLE OF THRUST VECTOR MUST BE SPECIFIED.
C  THE MODEL CAN ACCEPT REVERSE THRUST AND/OR DIFFERENTIAL
C  THRUST SCHEDULES.
C*********XENG,YENG AND ZENG=THE XYZ BODY AXIS MOMENT ARM
C         VECTORS (FEET)
C  THE REQUIRED INPUT PARAMETERS ARE
```

Figure 144 EASY XC-8A ACLS Model File For The Landing Simulation (Continued)

```
C*********THRLT=LEFT ENGINE THRUST(LBS.)
C*********THRRT=RIGHT ENGINE THRUST(LBS.)
C*********THETA=THE ANGLE OF THRUST APPLICATION IN THE XZ PLANE
C           (DEGREES)
C*********FSENG,BLENG,WLENG. THESE ARE THE THE FUSELAGE
C         STATION,BUTT LINE, AND WATER LINE LOCATIONS
C         OF THE RIGHT ENGINE THRUST VECTOR. (IN.)
C*********FSCG,BLCG,WLCG. THESE ARE THE FUSELAGE STATION,
C         BUTT LINE AND WATER LINE LOCATIONS OF THE C.G..(IN.)
      RPD=.01745329
      FLPRAD=FLAPS*RPD
      THRAD=THETA*RPD
      COSINE=COS(THRAD)
      SINE=SIN(THRAD)
      XENG=(FSCG-FSENG)/12.0
      YENG=(BLENG-BLCG)/12.0
      ZENG=(WLCG-WLENG)/12.0
      THDIF=THRLT-THRRT
      THRT=THRRT
      THLT=THRLT
      IF(THRT.LT.0.0) THRT=0.0
      IF(THLT.LT.0.0) THLT=0.0
      DIF=THLT-THRT
      FX1=(THRLT+THRRT)*COSINE
      FY1=(.002+.0183*FLPRAD)*DIF
      FZ1=-(THRLT+THRRT)*SINE
      T1=-.0176*(THRLT+THRRT)
      T2=(.0125+.1095*FLPRAD)*DIF
      TX1=T1+T2
      TY1=FX1*ZENG-FZ1*XENG
      TZ1=(.1575-.0322*FLPRAD)*THDIF
C*********************************************************************
C*********************************************************************
C THESE FORTRAN STATEMENTS DETERMINE THE BODY AXIS FORCES AND
C TORQUES PRODUCED WHEN A WING SKID TOUCHES THE GROUND.THE REQUIRED
C INPUT PARAMETERS ARE
C*********FSSKID,BLSKID,WLSKID ARE THE FUSELAGE STATION,BUTT LINE
C         AND WATER LINE LOCATIONS OF THE TIP OF THE UNDEFORMED
C         RIGHT SKID. IT IS ASSUMED THAT THE SKIDS ARE AT SYMMETRIC
C         LOCATIONS W.R.T. THE X-Z PLANE. (IN.)
C*********FSCG,BLCG,WLCG. THESE ARE THE FUSELAGE STATION,            WING
C         BUTT LINE AND WATER LINE LOCATIONS OF THE C.G.. (IN.)      SKID
C*********SPLOW=LOW SPRING CONSTANT OF THE SKID (I.E. NORMAL SPRING
C         RATE).(LBS/FT)
C*********SPHIGH=HIGH SPRING CONSTANT WHICH REPRESENTS THE SPRING
C         CONSTANT OF THE BOTTOMED OUT SKID.(LBS/FT)
C*********DEFMX=THE MAXIMIUM NORMAL DEFLECTION OF THE SKID.(FT)
C         DEFLECTIONS BEYOND DEFMX ARE CONTROLLED BY THE
C         HIGHER SPRING RATE.
```

Figure 144 EASY XC-8A ACLS Model File For The Landing Simulation (Continued)

```
C*********SKIDMU=THE COEFFICIENT OF DYNAMIC FRICTION BETWEEN
C         THE SKID AND THE GROUND.
C*********SKID-THE SKID IS ELIMINATED FROM THE CALCULATIONS
C         WHEN SKID=0.0.
      FX2=0.0
      FY2=0.0
      FZ2=0.0
      TX2=0.0
      TY2=0.0
      TZ2=0.0
      COMP=0.0
      XFORCE=0.0
      ZFORCE=0.0
      IF(SKID.EQ.0.0) GO TO 11
      XSKID=(FSCG-FSSKID)/12.0
      YSKID=(BLSKID-BLCG)/12.0
      ZSKID=(WLCG-WLSKID)/12.0
      IF(ROLSG.LT.0.0) YSKID=-YSKID
      RPD=.01745329
      CP=COS(PITSG*RPD)
      SP=SIN(PITSG*RPD)
      CR=COS(ROLSG*RPD)
      SR=SIN(ROLSG*RPD)
      CY=COS(YAWSG*RPD)                                              WING
      SY=SIN(YAWSG*RPD)                                              SKID
C*********HGT IS THE HEIGHT OF THE SKID ABOVE THE GROUND. IF HGT
C         IS NEGATIVE THE SKID SPRING IS COMPRESSED.
      HGT=+XSKID*SP-YSKID*SR*CP-ZSKID*CR*CP+ALTSG
      IF(HGT.GE.0.0) GO TO 11
      COMP=-HGT/(SQRT(1.0-(SR*CP)**2.))
C*********COMP IS THE AMOUNT THAT THE SKID SPRING IS COMPRESSED.
C         COMP DOES NOT EQUAL HGT BECAUSE THE AIRCRAFT IS ROTATED
C         W.R.T. THE EARTH. THE DENOMINATOR OF THIS ASSIGNMENT
C         STATEMENT IS THE COSINE OF THE ANGLE WHICH THE SKID
C         MAKES FROM VERTICAL.
      HIGH=SPHIGH
      IF(COMP.LT.DEFMX) HIGH=SPLOW
C         THE EARTH AXIS FORCES PRODUCED BY THE COMPRESSION OF
C         THE SKID SPRING ARE DETERMINED BELOW.
      ZFORCE=-(DEFMX*SPLOW+(COMP-DEFMX)*HIGH)
      XFORCE=SKIDMU*ZFORCE
      IF(IERR.NE.1) GO TO 211
      WRITE(6,210)
 210  FORMAT(10X,35HA WING SKID IS TOUCHING THE GROUND.)
 211  CONTINUE
C         THE BODY AXIS FORCES ARE CALCULATED BELOW.
      FX2=XFORCE*CP*CY-ZFORCE*SP
      FY2=XFORCE*(SR*SP*CY-CR*SY)+ZFORCE*SR*CP
      FZ2=XFORCE*(CR*SP*CY+SR*SY)+ZFORCE*CR*CP
```

Figure 144 EASY XC-8A ACLS Model File For The Landing Simulation (Continued)

```
C         THE BODY AXIS TORQUES ARE CALCULATED BELOW.
      TX2=YSKID*FZ2-ZSKID*FY2
      TY2=ZSKID*FX2-XSKID*FZ2
      TZ2=XSKID*FY2-YSKID*FX2
   11 CONTINUE
C THE NEXT 6 STATEMENTS SUM THE FORCES AND MOMENTS PRODUCED
C BY THE ENGINES AND SKIDS.
      FX1S2=FX1+FX2
      FY1S2=FY1+FY2
      FZ1S2=FZ1+FZ2
      TX1S2=TX1+TX2
      TY1S2=TY1+TY2
      TZ1S2=TZ1+TZ2
C*********************************************************************
C*********************************************************************
C THESE FORTRAN STATEMENTS DETERMINE THE AERODYNAMIC
C DERIVATIVES AND COEFFICIENTS USED IN COMPONENTS OL AND DL.
C THE CALCULATED DERIVATIVES AND COEFFICIENTS ARE ALL WITH
C RESPECT TO THE STABILITY AXIS AND ARE ALL NONDIMENSIONAL
C VALUES. THE MOMENT TERMS ARE FOR A C.G. LOCATION AT 40
C PERCENT MAC. XP1OL CORRECTS THE DATA FOR A DIFFERENT
C C.G. LOCATION.
C THE XC-8A AERODYNAMIC MODEL WAS OBTAINED FROM G.KURYLOWICH
C AIR FORCE FLIGHT DYNAMICS LABORATORY.
C         *********************************************
C         * THE FOLLOWING VARIABLES AND PARAMETERS    *
C         * ARE NEEDED TO DETERMINE THE AERODYNAMIC   *
C         * COEFFICIENTS AND DERIVATIVES. THESE ARE   *
C         * NOT EASY STANDARD COMPONENT VARIABLES OR  *
C         * PARAMETERS.                               *
C         *   TRUNK-  THE TRUNK IS DEFLATED WHEN      *
C         *         TRUNK=0.0, AND FULLY INFLATED     *
C         *         WHEN TRUNK=1.0.                   *
C         *   CGRAV-  THE LOCATION OF THE C.G.. IT    *
C         *         IS  INPUT AS A PERCENT MAC.       *
C         *   FLAPS-  THE ANGULAR DEFLECTION OF THE   *
C         *         FLAPS,INPUT IN DEGREES.           *
C         *   SPOIL-  THE FRACTION OF THE TOTAL       *
C         *         SPOILER DEFLECTION. THE RANGE     *
C         *         OF SPOIL IS FROM 0.0 TO 1.0. THE  *
C         *         SPOILERS ARE FULLY DEPLOYED WHEN  *
C         *         SPOIL=1.0, AND FULLY RETRACTED    *
C         *         WHEN SPOIL=0.0.                   *
C         *********************************************
C THIS NEXT GROUPING OF FORTRAN STATEMENTS EVALUATES
C FREQUENTLY USED EXPRESSIONS PRESENT IN THE AERODYNAMIC
C EQUATIONS.
C*********CTRT IS THE RIGHT ENGINE COEFFICIENT OF THRUST
C*********CTLT IS THE LEFT ENGINE COEFFICIENT OF THRUST.
```

Figure 144 EASY XC-8A ACLS Model File For The Landing Simulation (Continued)

```
C*********CTTOT IS THE TOTAL COEFFICIENT OF THRUST.
      CTRT=THRRT/QS VA
      CTLT=THRLT/QS VA
      CTTOT=CTRT+CTLT
C*********IF CTTOT IS NEGATIVE,CTTOT IS THEN SET TO ZERO.
      IF(CTTOT.LT.0.0) CTTOT=0.0
      CTT2=CTTOT**2.
      RPD=.01745329
      ALRAD=AL VA*RPD
      BERAD=BE VA*RPD
      COS2BE=(COS(BERAD))**2.
      SIN2BE=(SIN(BERAD))**2.
      BEABS=ABS(BERAD)
      FFBETA=ABS(1.-.955*(BEABS-.523))
      IF(BEABS.LE..523) FFBETA=1.0
      FLPRAD=FLAPS*RPD
      FLP2=FLPRAD**2.
      ABZF=(WLCG-160.)/12.0
      ABZC=(WLCG-74.6)/12.0
      PIH=3.141592654/2.0
C AERODYNAMIC CONTRIBUTIONS FOR THE APUS,DIFFUSERS AND CUSHION.
C ( A 40 PERCENT C.G. LOCATION WAS USED FOR ALL TERMS.)
      ABCL=1.59/QC VA
      IF(ALTSG.LE.11.0) ABCL=0.0
      ABSB=ABS(SIN(BE VA))
      CDMON=5.19/VT VA
C CUSHION RETRACTED INCREMENTS OGE FANS CLOSED
      RETCD=.004
      RETCY=0.0
      RETCM=-.004*(ABZF-7.9*ALRAD)/C  OL
      RETCLL=0.0
      RETCN=0.0
C CUSHION EXTENDED INCREMENTS FAN CLOSED
      EXTCD=.02
      EXTCY=-.05*ABSB
      EXTCM=-.02*(ABZC-5.54*ALRAD)/C  OL
      EXTCLL=-EXTCY*ABZC/B  DL
      EXTCN=EXTCY*5.54/B  DL
C CUSHION EXTENDED INCREMENTS WITH FANS OPERATIONAL
      EMCD=CDMON+ABCL*ALRAD
      EMCY=-CDMON
      EMCM=-CDMON*(ABZF-5.6*ALRAD)/C  OL+ABCL*5.54/C  OL
      EMCLL=-EMCY*ABZF/B  DL
      EMCN=EMCY*5.6/B  DL
C * * * * * * * * * * * * * * * * * * * * * * * * * * *
C
C AERODYNAMIC LIFT DERIVATIVES AND COEFFICIENTS
C
C********** BASIC LIFT COEFFICIENT **********
```

Figure 144 EASY XC-8A ACLS Model File For The Landing Simulation (Continued)

```
      Z1=(.35+2.75*FLPRAD-.616*FLP2+(.15+1.865*FLPRAD
     1-.821*FLP2)*CTTOT+(5.27+.72*CTTOT)*ALRAD)
     2*COS2BE
      ZO OL=-Z1
C********** ELEVATOR DERIVATIVE **********
      ZDEOL=-.566*COS2BE
C********** SPOILER DERIVATIVE **********
      ZSPOL=-(-.25-.573*FLPRAD)
      SPOOL=SPOIL/RPD
C        SPOIL IS DIVIDED BY RPD TO NEGATE THE EFFECTS
C        OF THE AUTOMATIC DEGREE TO RADIAN CONVERSION.
C********** TRUNK COEFFICIENT **********
      ZTROL=-TRUNK*ABCL
C * * * * * * * * * * * * * * * * * * * * * * * * * *
C
C AERODYNAMIC DRAG DERIVATIVES AND COEFFICIENTS
C
C********** BASIC DRAG COEFFICIENT **********
      CL2=(ZO OL+ZDEOL*ELEOL*RPD+ZTROL+ZSPOL*SPOOL*RPD)**2.
      X1=(.026+.0394*CL2+(.115+.46*CTTOT-.12*CTT2
     1)*FLPRAD)*COS2BE
      IF(X1.LT.0.0) X1=0.0
C          MODIFICATION OF BASIC DRAG COEFFICIENT
C          WHEN VT VA IS LESS THAN 70.0 FPS.
      X2=0.0
      IF(VT VA.GT.70.0) GO TO 15
      X2=(.229*FLPRAD+.176*FLP2)*CTTOT
      B=1.0-(VT VA-50.0)/20.0
      IF(50.0.LE.VT VA.AND.VT VA.LE.70.0) X2=X2*B
   15 CONTINUE
      XO OL=-X1-X2
C********** TRUNK COEFFICIENT **********
      XTROL=-(RETCD+TRUNK*(EXTCD+EMCD))
C********** SPOILER DERIVATIVE **********
      XSPOL=(-.0125-(.1*FLPRAD))
C * * * * * * * * * * * * * * * * * * * * * * * * * *
C
C AERODYNAMIC SIDEFORCE DERIVATIVES AND COEFFICIENTS
C
C********** BETA DERIVATIVE **********
      IF(BEABS.GT..436) GO TO 18
      A=-.8-.515*FLPRAD
      B=-.1-.702*FLPRAD
      C=-.57-1.89*FLPRAD+.82*FLP2
      D=.573-1.64*FLPRAD
      IF(FLPRAD.GE..349) D=-1.145
      Y1=A+B*CTTOT+(C+D*CTTOT)*ALRAD
      GO TO 19
   18 A=.08+.225*FLPRAD
```

Figure 144 EASY XC-8A ACLS Model File For The Landing Simulation (Continued)

```
      B=(.025+.322*FLPRAD)*CTTOT-.079*BERAD
      IF(ABS(CTTOT).LT..1) B=B+.079*BERAD
      IF(BERAD-PIH) 500,501,501
 500  Y1=-(1.0/BEABS)*(1.38*SIN2BE+A*(1.0
     1-(BEABS-.436)/1.134)+B)
      GO TO 19
 501  Y1=-(1.0/BEABS)*(1.38*SIN2BE+A*(PIH-BEABS)/
     1   1.134+B)
  19  YB DL=Y1
C********** AILERION DERIVATIVE **********
      YDADL=.0277*(.0473+.0545*FLPRAD)*FFBETA
C********** RUDDER DERIVATIVE AND EFFECTIVENESS **********
      YDRDL=.407
      YBRDL=FFBETA
C********** ROLL RATE DERIVATIVE **********
      YP DL=-.06
C********** YAW RATE DERIVATIVE **********
      YR DL=.704
C********** TRUNK COEFFICIENT **********
      YTRDL=RETCY+TRUNK*(EXTCY+EMCY)
C * * * * * * * * * * * * * * * * * * * * * * * * *
C
C AERODYNAMIC PITCHING MOMENT DERIVATIVES AND COEFFICIENTS
C
C********** BASIC PITCHING MOMENT COEFFICIENT **********
      IF(CTTOT.GE.1.0) GO TO 20
      A=.2-.2*CTTOT
      B=.251-.036*CTTOT
      C=-.245*CTTOT
      D=-3.69+3.616*CTTOT
      E=-1.645-5.375*CTTOT
      F=4.71*CTTOT
      GO TO 21
  20  A=.2*(CTTOT-1.0)
      B=.215+1.002*(1.0-CTTOT)
      C=-.49+.245*CTTOT
      D=-.074+1.566*(1.0-CTTOT)
      E=-14.04+7.02*CTTOT
      F=9.42-4.71*CTTOT
  21  M1=A+B*FLPRAD+C*FLP2+(D+E*FLPRAD+F*FLP2)
     1*(ALRAD+.1745)**2.
      MO OL=M1
C********** PITCH RATE DERIVATIVE **********
      MQ OL=-38.4
C********** ALPHA DOT DERIVATIVE **********
      MADOL=-7.-5.*CTTOT
      IF(CTTOT.GE.1.0) MADOL=-12.0
C********** TRUNK COEFFICIENT **********
      MTROL=(RETCM+EXTCM+EMCM)*TRUNK
```

Figure 144 EASY XC-8A ACLS Model File For The Landing Simulation (Continued)

```
C********** ELEVATOR DERIVATIVE **********
      MDEOL=-2.43*COS2BE
C********** SPOILER DERIVATIVE
      MSPOL=(.02+.1145*FLPRAD)
C * * * * * * * * * * * * * * * * * * * * * * * * *
C
C AERODYNAMIC ROLL MOMENT DERIVATIVES AND COEFFICIENTS
C
C********** BETA DERIVATIVE **********
      IF(BEABS.GT..436) GO TO 31
      L1=(-.164+.043*FLPRAD+(.014+.1575*FLPRAD-.0924*FLP2)
     1*CTTOT+(.172+(.1035-.0186*FLPRAD-.082*FLP2)
     2*CTTOT)*ALRAD)
      GO TO 32
   31 A=.01875*FLPRAD
      B=(.0041+.0326*FLPRAD)*CTTOT
      IF(BEABS-PIH) 510,511,511
  510 L1=(1.0/BEABS)*(-.022-.235*SIN2BE*COS2BE+
     1A*(1.0-(BEABS-.436)/1.134)+B)
      GO TO 32
  511 L1=(1.0/BEABS)*(-.022-.235*SIN2BE*COS2BE+A
     1   *(PIH-BEABS)/1.134+B)
   32 LB DL=L1
C********** ROLL RATE DERIVATIVE **********
      LP DL=-.53-.1*FLPRAD+(.08-.0572*FLPRAD)*CTTOT
     1+.516*ALRAD
C**********YAW RATE DERIVATIVE **********
      LR DL=.15+.587*FLPRAD+.975*ALRAD
C********** AILERION DERIVATIVE **********
      LDADL=(.0473+.0545*FLPRAD)*FFBETA
C********** RUDDER DERIVATIVE AND EFFECTIVENESS PARAMETER **********
      LDRDL=.184*SIN(.265-ALRAD)
      LBRDL=FFBETA
C********** TRUNK COEFFICIENT **********
      LTRDL=RETCLL+TRUNK*(EXTCLL+EMCLL)
C * * * * * * * * * * * * * * * * * * * * * * * * *
C
C AERODYNAMIC YAW MOMENTS DERIVATIVES AND COEFFICIENTS
C
C********** BETA DERIVATIVE **********
      IF(BEABS.GT..436) GO TO 37
      A=.125+.0329*FLPRAD
      B=-.015-.0415*FLPRAD
      C=.057+.52*FLPRAD-.495*FLP2
      D=-.0573-.181*FLPRAD
      IF(FLPRAD.GE..349) D=-.1204+.1315*(FLPRAD-.349)
      N1=A+B*CTTOT+(C+D*CTTOT)*ALRAD
      GO TO 38
   37 A=.02475+.01435*FLPRAD
```

Figure 144 EASY XC-8A ACLS Model File For The Landing Simulation (Continued)

```
      B=(.006+.0236*FLPRAD)*CTTOT
      IF(BEABS-PIH) 520,521,521
 520  N1=(1.0/BEABS)*(.106*SIN2BE+A*(1.0
     1-(BEABS-.436)/1.134)-B)
      GO TO 38
 521  N1=(1.0/BEABS)*(.106*SIN2BE+A*(PIH-BEABS)/1.134-B)
  38  NB DL=N1
C********** YAW RATE DERIVATIVE **********
      NR DL=-.22-.06*FLPRAD-(.129+.308*FLPRAD)*ALRAD
C********** ROLL RATE DERIVATIVE **********
      NP DL=((-.1+.009*CTTOT)*FLPRAD)/.699+
     1(-.5-.145*CTTOT*(FLPRAD-.699)/.699)*ALRAD
C********** AILERION DERIVATIVE **********
      NDADL=-(.0467+.704*ALRAD)*LDADL
C********** RUDDER DERIVATIVE AND EFFECTIVENESS PARAMETER **********
      NDRDL=-.184*COS(.265-ALRAD)
      NBRDL=FFBETA
C********** TRUNK COEFFICIENT **********
      NTRDL=RETCN+TRUNK*(EXTCN+EMCN)
C XP10L CORRECTS THE AERODYNAMIC DATA FOR A C.G. LOCATION
C DIFFERENT FROM 40 PERCENT MAC USED IN DERIVING THE DATA.
      XP10L=C  OL*(.01*CGRAV-.4)
C*********************************************************************
C*********************************************************************
C  THE FOLLOWING FORTRAN STATEMENTS EVALUATE THE GROUND EFFECTS
C  FOR THE AERODYNAMIC MODEL. THE PROCEEDURE USED TO EVALUATE
C  GROUND EFFECTS WAS TAKEN FROM DATCOM SECTION 4.7 METHOD 1.
C  THE REQUIRED INPUT PARAMETERS ARE
C*********ROOTC=ROOT CHORD (CR) LENGTH (FEET)
C*********SH=HORIZONTAL STABILIZER AREA (SQUARE FEET)
C*********ASPECT=ASPECT RATIO
C*********FS25CR,WL25CR=THE FUSELAGE STATION AND WATER
C         LINE LOCATIONS OF .25 ROOT CHORD.
C*********FS75CR,WL75CR=THE FUSELAGE STATION AND WATER
C         LINE LOCATIONS OF THE .75 ROOT CHORD.
C*********FSTAIL,WLTAIL=FUSELAGE STATION AND WATER
C         LINE LOCATIONS OF THE HORIZONTAL STABILIZERS
C         AERODYNAMIC CENTER.
C  THE REQUIRED INPUT TABLES ARE
C*********TBCL0,TBCL20,TBCL40= THE TOTAL AIRPLANES COEFFICIENT
C         OF LIFT AT 0.0,20.0,40.0 DEG. FLAPS RESPECTIVELY.
C         THESE TABLES ARE FUNCTIONS OF ALPHA AND CTTOT.
C*********TBCLH=THE HORIZONTAL STABILIZER COEFFICIENTS OF
C         INPUT AS FUNCTIONS OF HORIZONTAL ALPHA AND ELEOL.
C*********TBX=THE X VARIABLES FROM DATCOM FIGURE 4.7.1-14.
C*********TBLLO=THE L/LO-1. VARIABLE FROM DATCOM FIGURE 4.7.1-15.
C*********TBDEL=THE DELTA DELTA CL VARIABLE FROM DATCOM
C         FIGURE 4.7.1-17 FOR SLOTTED FLAPS.
C  ASSUMPTIONS
```

Figure 144 EASY XC-8A ACLS Model File For The Landing Simulation (Continued)

```
C          1. SWEEP ANGLE OF THE QUARTER CHORD LINE
C             OF THE WING = 0.0 DEG..
C          2. THE DYNAMIC PRESSURE OF THE HORIZONTAL
C             STABILIZER EQUALS THE DYNAMIC PRESSURE
C             AT INFINITY.
C          3. THE WINGS ARE STRAIGHT AND PERPENDICULAR
C             TO THE X-Z PLANE. THUS H.75CR=H.75B/2.
C             SEE DATCOM FIGURE 4.7.1-14.
C          4. THE MAIN WINGS AERODYNAMIC CENTER IS ASSUMED
C             TO BE AT .25C..
C  IF THE AIRCRAFTS ALTITUDE IS GREATER THAN TWICE THE
C  WING SPAN THE GROUND EFFECTS ARE SKIPPED.
      DELCLGR=0.0
      DELCDLG=0.0
      DELCMG=0.0
      IF(ALTSG.GT.2.*B  DL) GO TO 777
C  THE NEXT GROUPING OF FORTRAN STATEMENTS EVALUATES
C  FREQUENTLY USED VARIABLES IN THE GROUND EFFECTS
C  CALCULATIONS.
C
      SRATIO=SH/S  VA
      EPSILON=AL VA*(.24-.0015*FLAPS)
      TAILA=AL VA-EPSILON
      X75CR=(FSCG-FS75CR)/12.0
      Z75CR=(WLCG-WL75CR)/12.0
      X25CR=(FSCG-FS25CR)/12.0
      Z25CR=(WLCG-WL25CR)/12.0
      XTAIL=(FSCG-FSTAIL)/12.0
      ZTAIL=(WLCG-WLTAIL)/12.0
      RPD=.01745329
      SP=SIN(PITSG*RPD)
      CR=COS(ROLSG*RPD)
      CP=COS(PITSG*RPD)
      CRCP=CR*CP
      H75CR=X75CR*SP-Z75CR*CRCP+ALTSG
      IF(H75CR.LE.11.25) GO TO 777
      H25CR=X25CR*SP-Z25CR*CRCP+ALTSG
      HTAIL=XTAIL*SP-ZTAIL*CRCP+ALTSG
      H75=2.*H75CR/B  DL
      H25=H25CR/ROOTC
      CRB=ROOTC/B  DL
C  THE NEXT GROUPING OF FORTRAN STATEMENTS EVALUATES
C  THE TOTAL AIRCRAFTS COEFFICIENT OF LIFT BY TABLE
C  LOOK UP ROUTINES INPUT AS FUNCTIONS OF ALPHA AND CTTOT.
C  THERE ARE 3 INPUT TABLES FOR FLAP SETTINGS OF 0.0,20.0
C  AND 40.0 DEGREES. A LINEAR INTERPELATION IS MADE
C  FOR FLAP SETTINGS BETWEEN THESE VALUES.
      I=0
      IF(FLAPS.GE.20.) GO TO 2
```

Figure 144 EASY XC-8A ACLS Model File For The Landing Simulation (Continued)

```
      T=FLAPS/20.0
      GO TO 4
    2 T=(FLAPS-20.)/20.
    4 A=AL VA
    5 I=I+1
      IF(FLAPS.GE.20.) GO TO 14
      COEL1=TBLU2(A,CTTOT,TBCLO(7),TBCLO(4),TBCLO(17)
     1,1,1,10,3,10,3)
      COEL2=TBLU2(A,CTTOT,TBCL20(7),TBCL20(4),TBCL20(17)
     1,1,1,10,3,10,3)
      GO TO 150
   14 COEL1=TBLU2(A,CTTOT,TBCL20(7),TBCL20(4),TBCL20(17)
     1,1,1,10,3,10,3)
      COEL2=TBLU2(A,CTTOT,TBCL40(7),TBCL40(4),TBCL40(17)
     1,1,1,10,3,10,3)
  150 CL=(COEL2-COEL1)*T+COEL1
      GO TO (10,200,30,40) I
C  CALCULATION OF THE GROUND EFFECTS UPDATE FOR
C  THE LIFT COEFFICIENT.
   10 CLWBH=CL
      CLH=TBLU2(TAILA,ELEOL,TBCLH(7),TBCLH(4),TBCLH(15)
     1,1,1,8,3,8,3)
C*********CLFWB=IS THE WING-BODY LIFT COEFFICIENT INCLUDING
C         FLAP EFFECTS, OUT OF GROUND EFFECT.
      CLFWB=CLWBH-CLH*SRATIO
      X=TBLU1(H75,TBX(4),TBX(15),1,11)
      IF(X.LT.0.0) X=0.0
      ALUP=AL VA+.5
      ALDN=AL VA-.5
      EUP=ALUP*(.24-.0015*FLAPS)
      EDN=ALDN*(.24-.0015*FLAPS)
      AHUP=ALUP-EUP
      AHDN=ALDN-EDN
      A=ALUP
      GO TO 5
  200 CLUP=CL
      A=ALDN
      GO TO 5
   30 CLDN=CL
      CLHUP=TBLU2(AHUP,ELEOL,TBCLH(7),TBCLH(4),TBCLH(15)
     1,1,1,8,3,8,3)
      CLHDN=TBLU2(AHDN,ELEOL,TBCLH(7),TBCLH(4),TBCLH(15)
     1,1,1,8,3,8,3)
C*********CLAWB=IS THE WING-BODY LIFT CURVE SLOPE,PER DEGREE
C         OUT OF GROUND EFFECT. IT IS EVALUATED BY CALCULATING
C         CLFWB FOR AL VA+.5 AND AL VA-.5 DEGREES. CLAWB IS
C         ASSIGNED THE AVERAGE SLOPE CALCULATED FOR THIS
C         ONE DEGREE CHANGE IN ALPHA.
      CLAWB=CLUP-CLDN-SRATIO*(CLHUP-CLHDN)
```

Figure 144 EASY XC-8A ACLS Model File For The Landing Simulation (Continued)

```
      CKCL=CLWBH*9.1196
      OLL=TBLU2(H25,CKCL,TBLLO(7),TBLLO(4),TBLLO(19)
     1,1,1,-12,3,12,3)
      R=SQRT(1.+H75**2.)-H75
      DELDEL=TBLU1(H25,TBDEL(4),TBDEL(9),1,5)
      IF(DELDEL.GT.0.0) DELDEL=0.0
      CK11=9.12/ASPECT+7.16*CRB
      CK12=ASPECT*CRB/2.
      CK13=(FLAPS/50.)**2.
C*********DELAL=THE CHANGE IN ALPHA AT THE PRESENT
C          ALTITUDE.
      DELAL=-CK11*CLFWB*X-(CK12/CLAWB)*OLL*CLFWB*R-CK13*DELDEL/CLAWB
      IF(DELAL.GT.0.0) DELAL=0.0
      ALGR=AL VA-DELAL
      A=ALGR
      GO TO 5
   40 CL1=CL
      DELCLGR=CLWBH-CL1
      ZO OL=ZO OL+DELCLGR
C  CALCULATION OF THE GROUND EFFECT UPDATE FOR
C  THE MOMENT M COEFFICIENT.
      FLAPNO=TBLU2(AL VA,CTTOT,TBCLO(7),TBCLO(4),TBCLO(17)
     1,1,1,10,3,10,3)
C*********CLWB=THE WING-BODY LIFT COEFFICIENT, FLAPS RETRACTED,
C          OUT OF GROUND EFFECT.
      CLWB=FLAPNO-SRATIO*CLH
C*********DELCLF=THE CHANGE IN LIFT COEFFICIENT DUE TO FLAPS,
C          OUT OF GROUND EFFECT.
      DELCLF=CLWBH-FLAPNO
C*********BEFF=THE EFFECTIVE WING SPAN.
      BEFF=(CLWB+DELCLF)/((CLWB/74.4)+(DELCLF/58.8))
      BEFF2=BEFF**2.
      SN1=BEFF2+4.*(HTAIL-H25CR)**2.
      SN2=BEFF2+4.*(HTAIL+H25CR)**2.
      DELE=EPSILON*SN1/SN2
      HAL=TAILA+DELE
      CLHGR=TBLU2(HAL,ELEOL,TBCLH(7),TBCLH(4),TBCLH(15)
     1,1,1,8,3,8,3)
      DELCLHG=CLHGR-CLH
C*********DELCMHG=THE CHANGE IN PITCHING MOMENT COEFFICIENT
C          PRODUCED BY THE HORIZONTAL STABILIZER. A .4MAC
C          C.G. LOCATION WAS ASSUMED.
      DELCMHG=-DELCLHG*SRATIO*4.304
      DCLWBG=DELCLGR+DELCLHG*SRATIO
C*********DCMUBG=THE CHANGE IN PITCHING MOMENT FOR THE
C          WING. A .4C C.G. LOCATION WAS ASSUMED.
      DCMWBG=-.15*DCLWBG
      DELCMG=DCMWBG+DELCMHG
      MO OL=MO OL+DELCMG
```

Figure 144 EASY XC-8A ACLS Model File For The Landing Simulation (Continued)

```
C CALCULATION OF THE GROUND EFFECTS UPDATE FOR
C  THE DRAG COEFFICIENT.
      SIGMA=EXP(-2.48*H75**.768)
      DELCDLG=SIGMA*(CLFWB**2.)/(ASPECT*3.14159)
      XO OL=XO OL+DELCDLG
 777  CONTINUE
C*********************************************************************
C*********************************************************************
LOCATION=80     FL      INPUTS=SG,VA(MAC=AMN)
LOCATION=76     FN      INPUTS=FL,TS(PT=P,2)
FORTRAN STATEMENTS
C   THE FOLLOWING STATEMENT CONVERTS THE LBS/SEC FLOW RATE TO
C   LBS/MIN. THIS VALUE IS THEN DOUBLED TO ACCOUNT FOR THE TWO
C   FANS IN THE SYSTEM.  (I.E. 60*2=120)
      WTRTS=120.*W2 FN
LOCATION=17     TS      INPUTS=SG,FL(PAM=PA),FN(T,2=TTR)
LOCATION=14     S2      INPUTS=
TS(FXT=FX,2,FYT=FY,2,FZT=FZ,2,TXT=TX,2,TYT=TY,2,TZT=TZ,2)
LOCATION=34     OL      INPUTS=VA,S2
LOCATION= 1     DL      INPUTS=VA,OL,S2
LOCATION=40     SG      INPUTS=OL,DL
FORTRAN STATEMENTS
      DPR=57.29578
      XZ2=IXZSG**2.
      XYZ=IXXSG*IYYSG*IZZSG                      CODING FOR
      QD=DPR*TY2OL*(IXXSG*IZZSG-XZ2)/             PSEUDO-INTEGRAL
     1   (XYZ-XZ2*IYYSG)                          CONTROLLER IT2
      FINIT2=QD-2.3
      RPD=.01745329
      SP=SIN(PITSG*RPD)
      SR=SIN(ROLSG*RPD)
      CP=COS(PITSG*RPD)
      CR=COS(ROLSG*RPD)
      ACELCG=(FX2OL*SP-FY2DL*SR*CP-FZ2OL*CR*CP)/(32.174*MA1OL)-1.
      PCUSH=(PC TS-PAMFL)*144.0
      PTRUNK=(PT TS-PAMFL)*144.0
C*********ACELCG=THE VERTICAL ACCELERATION OF THE C.G.. (G)
C*********PCUSH=GAGE CUSHIOH PRESSURE (PSFG)
C*********PTRUNK=GAGE TRUNK PRESSURE (PSFG)
LOCATION=103     IT2
END OF MODEL
PRINT
```

Figure 144 EASY XC-8A ACLS Model File For The Landing Simulation (Concluded)

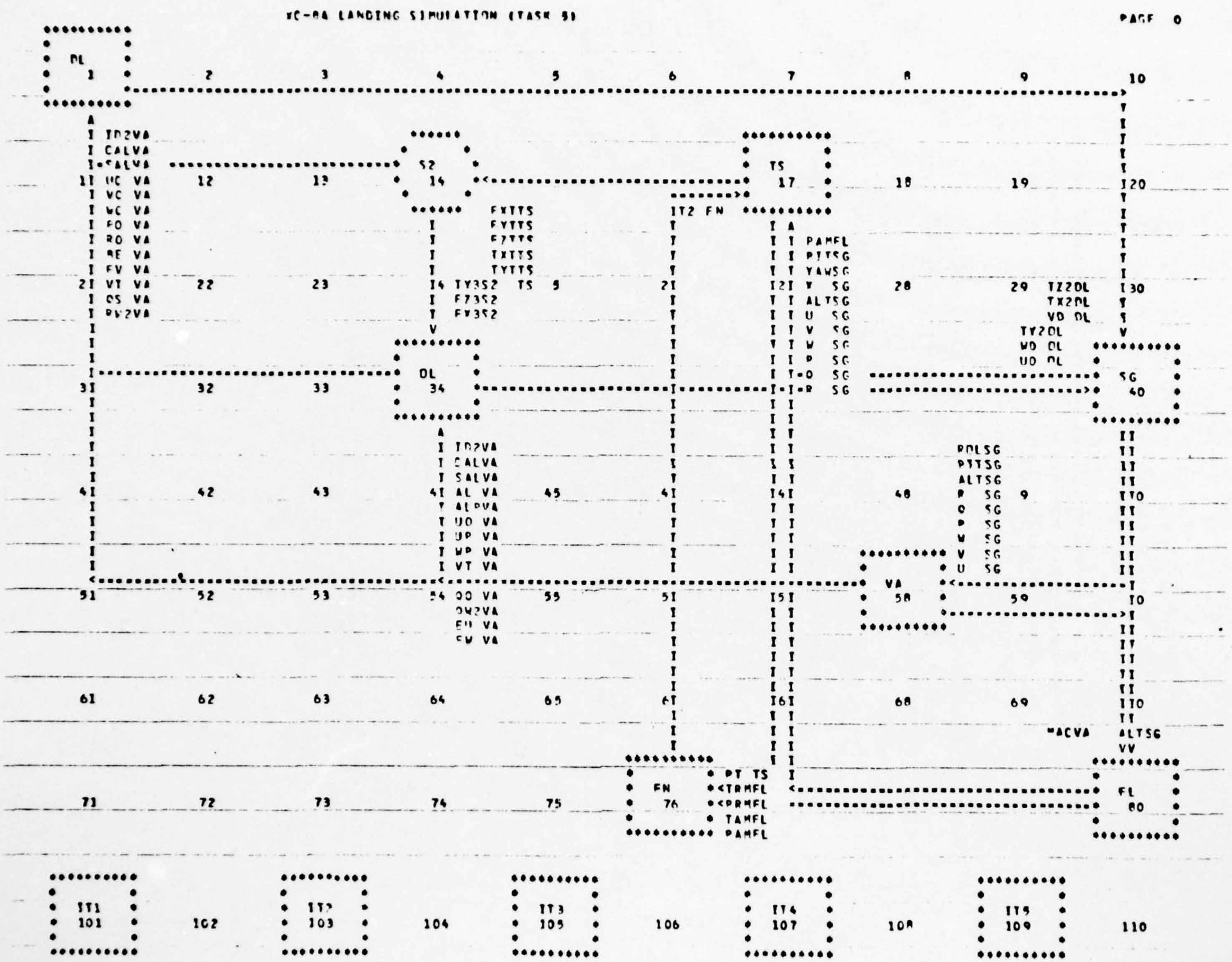


Figure 145 EASY XC-8A ACLS Model Schematic For The Landing Simulation

components FL, FN, TS and S2. FL calculates the stationary atmospheric temperature and pressure along with the ram temperature and pressure (assuming 100% efficiency) for a user specified day. These values are input into FN which is an inlet fan model implemented to simulate the ASP-10 fans. Only one fan component is implemented in the computer model. The output from this component is doubled using Fortran statements in the model file to account for the two ASP-10's fans in the XC-8A aircraft. The elastic trunk is modeled by component TS. The ambent air pressure input into TS is also calculated in component FL. Component S2 sums the body axis forces and moments produced by the elastic trunk TS, the two engines and the two wing skids. S2 then inputs these totals as external forces and moments into OL and DL.

The remaining standard components in the model file which have not been discussed in previous sections are IT1, IT2, IT3, IT4 and IT5. These are the psedo-intergral controllers implemented to initialize the model. For a discussion of model initialization using pseudo-intergral controllers see Appendix B. The model initialization performed by each of these controllers are described below:

- IT1 varies the engine thrust until the sink rate (or altitude rate) is -8.0 ft/sec.
- IT2 varies the elevator deflection until the rate of Q is 2.3 degrees/sec^2.
- IT3 varies the rudder deflection until the v velocity is zero.
- IT4 varies the aileron deflection until the p angular rate is zero.
- IT5 varies the trunk to cushion trim valve area (CAVTS) until the trunk pressure is 16.627 psia.

The input and output connections to these controllers are made by Fortran statements, which have been bracketed in Figure 144.

The wing skid forces and moments generated whenever a skid contacts the ground are determined by Fortran statements in the model file (See Figure 144). The

required input parameters are listed and defined at the beginning of the coding. This coding models the wing skid as a linear spring with two spring rates. The smaller spring rate (SPLOW) represents the compression of the spring at the bottom of the skid. The larger rate (SPHIGH) represents the compression of the skid when the spring is fully compressed. Whenever the wing skid touches the ground two forces are produced. There is a vertical force perpendicular to the runway and a tangential frictional force parallel to the runway. These two forces are resolved into body axis forces and moments and added to the body axis forces and moments produced by the engine. This total is then input into port 1 of component S2. The coding for the wing skid will also notify the user when a wing skid touches the ground by printing "A WING SKID IS TOUCHING THE GROUND" in the time history printout.

The variables ACELCG, VKNOTS, PCUSH, PTRUNK are defined in the model file using Fortran statements to allow a direct comparision with the fight test data. These values are respectively the vertical acceleration of the c.g. in g's, the airspeed in knots, the cushion pressure in psfg and the trunk pressure in psfg.

9.2.2 Analysis File

The analysis file for the XC-8A aircraft simulation is listed in Figure 146. The input data listed in this file for OL, DL, SG, VA, the ground effects and the aerodynamic coefficient and derivative coding was taken directly from the XC-8A aircraft simulation (Section XIII). The only changes which were made in these parameters reflect the shift in c.g. location from 40% for the aircraft simulation to 28% for the landing simulation. In addition to the aerodynamic data the analysis file includes data for the elastic trunk TS, the wing skid and the ASP-10 fan which model the ACS on the XC-8A aircraft.

The analysis file data for the XC-8A elastic trunk represents the Number 2 Flight Trunk for the XC-8A aircraft. For implementation into component TS the trunk was divided into three sets and 32 trunk elements with 16 elements per

```
TABLE=TBCL0,10,3
0.0,0.8,1.2
-4.0,-2.0,0.0,2.0,4.0,6.0,8.0,10.0,12.0,14.0
-.0154,.1523,.3378,.5153,.6988,.8734,1.0385
1.1809,1.3020,1.4271
-.0480,.1909,.4496,.7179,.9861,1.2423,1.4788
1.6915,1.8735,2.0650
-.0692,.2094,.4975,.7855,1.0790,1.3720,1.6264
1.8644,2.0840,2.2780
TABLE=TBCL20,10,3
0.0,0.8,1.2,
-4.0,-2.0,0.0,2.0,4.0,6.0,8.0,10.0,12.0,14.0
1.0055,1.1849,1.3435,1.4986,1.6568,1.8134
1.9626,2.0934,2.2089,2.3156
1.4464,1.7130,1.9513,2.1828,2.3979,2.6214
2.8429,3.0386,3.2056,3.3703
1.6034,1.8913,2.1564,2.3988,2.6470,2.8904
3.1287,3.3488,3.5450,3.7086
TABLE=TBCL40,10,3
0.0,0.8,1.2
-4.0,-2.0,0.0,2.0,4.0,6.0,8.0,10.0,12.0,14.0
1.6639,1.8323,1.9766,2.1411,2.3094,2.4749
2.5999,2.7305,2.8406,2.9339
2.4030,2.6580,2.8879,3.1195,3.3492,3.5917
3.7955,3.9699,4.1346,4.2591
2.6590,2.9325,3.1828,3.4399,3.7065,3.9480,4.1946
4.4070,4.5844,4.7244
TABLE=TBX,11
0.0,.1,.2,.3,.4,.5,.6,.7,.8,.9,1.0
1.0334,.7145,.5360,.4176,.3239,.2586,.2080
.1669,.1352,.1087,.0899
TABLE=TBCLH,8,3
-25.0,0.0,15.0
-18.0,-16.0,-14.0,8.0,10.0,12.0,14.0,16.0
-1.6755,-1.6289,-1.5699,-.3587,-.2409
-.1127,.00755,.1004
-1.0937,-1.0272,-.9288,.4601,.5818
.7110,.7566,.6414
-.6462,-.5369,-.4162,1.0091,1.1333,1.0773
.9537,.8311
TABLE=TBLL0,12,3
10.0,18.0,24.0
.2,.4,.6,.8,1.0,1.2,1.4,1.6,1.8,2.0,2.2,2.4
.2118,.06113,.00509,-.01720,-.02592,-.02780,-.02880,-.02938
-.02902,-.02856,-.02814,-.02824
-.05664,-.09657,-.1098,-.1063,-.09462,-.08550,-.07764,-.07006
-.06439,-.06119,-.05832,-.05740
-.1942,-.1980,-.1855,-.1648,-.1426,-.1250,-.1114
-.09891,-.08937,-.08117,-.07524,-.07178
```

Figure 146 EASY XC-8A ACLS Analysis File For The Landing Simulation

```
TABLE=TBDEL,5
.2,.4,.6,.8,1.0
-.09898,-.06246,-.03709,-.01953,-.005234
TABLE=ABLTS,9
71.9,0.0,71.9,.05,0.0,.5
47.1,10.0,57.1,.12,0.0,.5
44.2,15.0,56.8,.17,0.0,.3
TABLE=XYZTS,32
148.91,1.6,62.5,80.0
147.81,4.5,62.5,60.0
145.81,6.9,62.5,40.0
142.41,8.7,62.5,15.0
124.56,9.0,62.5,0.0
93.56,9.0,62.5,0.0
62.56,9.0,62.5,0.0
29.41,9.0,62.5,0.0
5.91,9.0,62.5,0.0
-17.84,9.0,62.5,0.0
-41.34,9.0,62.5,0.0
-65.09,9.0,62.5,0.0
-83.79,8.8,62.5,-12.5
-87.09,7.4,62.5,-35.0
-89.49,4.8,62.5,-57.5
-90.79,1.6,62.5,-80.0
TABLE=DM TS,16
20.0,.097,20.0,.097,20.0,.097,30.0,.097
31.0,.097,31.0,.097,31.0,.097,34.0,.097
13.25,.097,34.0,.097,13.25,.097,34.0,.097
22.5,.097,22.5,.097,25.0,.097,20.0,.097
TABLE=IALTS,32
1,.02189,21.0,23.0
1,.02189,21.0,23.0
2,.01911,22.5,14.0
2,.01911,22.5,14.0
3,.01911,22.25,14.0
3,.01911,22.25,14.0
3,.01911,22.25,14.0
3,.0025,19.5,21.0
3,.01911,23.5,13.0
3,.0025,19.5,21.0
3,.01911,23.5,13.0
3,.0025,19.5,21.0
2,.01911,22.5,14.0
2,.01911,22.5,14.0
1,.02189,21.0,23.0
1,.02189,21.0,23.0
TABLE=ENDTS,2
9.0,.45,9.0,.45
TABLE=SPHTS,8,3
```

Figure 146 EASY XC-8A ACLS Analysis File For The Landing Simulation (Continued)

```
1,2,3
0.0,33.,66.,100.,150.,200.,300.,500.
0.0,.2288,.6903,1.0079,1.1108,1.1963,1.3127,1.4880
0.0,.7809,1.8073,2.1367,2.2924,2.4125,2.5653,2.76
0.0,.9804,2.2110,2.5446,2.7195,2.8521,3.0180,3.2197
TABLE=STHTS,8,3
1,2,3
0.0,33.,66.,100.,150.,200.,300.,500.
0.0,.9804,2.2110,2.5446,2.7195,2.8521,3.0180,3.2197
0.0,.9804,2.2110,2.5446,2.7195,2.8521,3.0180,3.2197
0.0,.0135,.02702,.04094,.0614,.08187,.1228,.16374
TABLE=PM TS,14
7,0.0,0.0,.9
8,.8,5.0,.6
9,0.0,0.0,.8
10,.8,5.0,.6
11,0.0,0.0,.8
12,.8,5.0,.6
13,0.0,0.0,.9
TABLE=RELTS,2
0.0,50.0
0.0,0.0
TABLE=BWTTS,4
344.1,160.0,344.1,160.0,0,0,0,0
TABLE=FANFN,16,3
5260.,5880.,6370.
1.0,1.051,1.075,1.100,1.126,1.150,1.175,1.195
1.201,1.221,1.242,1.251,1.279,1.289,1.300,1.339
62.56,60.64,59.11,57.0,54.14,50.66,45.82,39.37
-.203,-1.444,-2.615,-3.158,-4.715,-5.258,-5.897,-8.153
69.18,67.8,66.81,65.22,63.39,61.3,58.74,56.25,55.48
51.73,45.52,-.275,-1.939,-2.483,-3.05,-5.353
74.87,73.67,72.99,72.09,70.92,69.55,67.97,66.25
65.77,63.56,60.81,59.16,50.28,-.231,-.763,-2.923
TABLE=STAFN,17
1.0,1.021,1.041,1.061,1.081,1.100,1.120,1.140
1.16,1.181,1.201,1.222,1.241,1.260,1.281,1.30,1.312
0.0,7.0,12.54,17.19,21.23,25.01,28.45,31.54
34.57,37.42,40.05,42.95,45.31,47.82,50.36,52.82,54.17
TABLE,ELEV,6
27.9,28.75,29.3,29.6,30.35,31.0
-10.408,0.5,-5.7,-5.5,-2.6,-3.2
TABLE,SPOLER,4
0.0,28.1,28.8,40.0
0.0,0.0,1.0,1.0
PARAMETER VALUES
NUIFN=.2,NUFFN=.87
CORFN=1.0,RPMFN=6186.
FSENG=197.0,BLENG=183.0,WLENG=191.0
```

Figure 146 EASY XC-8A ACLS Analysis File For The Landing Simulation (Continued)

```
FSCG=343.41,BLCG=0.0,WLCG=160.0
FS75CR=406.81,WL75CR=206.06,FS25CR=336.79
WL25CR=206.06,FSTAIL=889.65,WLTAIL=403.84
ROOTC=11.67,ASPECT=9.75,SH=233.0
THETA=0.0,TRUNK=1.0
CGRAV=28.0,FLAPS=20.0
ID1VA=3,VS VA=130.045,ALSVA=5.73
S  VA=945.0,IDGVA=6
WCUTS=0.0,TCUTS=560.0
ANETS=-16,CDGTS=1.0,CD1TS=.62
CD2TS=.2,TAUTS=.005,DMPTS=.02
EPCTS=1.
VU TS=6.0,PTMTS=3.0
SPBTS=0.0
MA1OL=1119.,C  OL=10.29,ISWOL=3
B  DL=96.0
IXXSG=183700.,IYYSG=228200.
IZZSG=365100.,IXZSG=29220.
FSSKID=340.0,BLSKID=432.5,WLSKID=90.0
SPLOW=2666.,SPHIGH=50000.,DEFMX=1.875
SKIDMU=.02,SKID=1.0
GKIIT1=10.0,GKIIT2=10.0,GKIIT3=10.0
GKIIT4=10.0,GKIIT5=10.0
INITIAL CONDITIONS
U  SG=129.3952,V  SG=0.0,W  SG=12.9838
P  SG=0.0,Q  SG=2.0,R  SG=0.0
ROLSG=0.0,PITSG=2.9,YAWSG=0.0
ALTSG=10.0,X  SG=0.0,Y  SG=0.0
PT TS=16.627,VT TS=850.29
PC TS=14.731,VC TS=139.47
FO IT1=130.,FO IT2=-3.0
FO IT3=1.0,FO IT4=0.0,FO IT5=119.
ERROR CONTROLS
U  SG=.001,V  SG=.0001,W  SG=.001
P  SG=.0001,Q  SG=.0001,R  SG=.0001
ROLSG=.0001,PITSG=.0001,YAWSG=.0001
ALTSG=.0001,X  SG=.0001,Y  SG=.0001
PT TS=.0001,VT TS=.001
PC TS=.0001,VC TS=.001
FO IT1=.0001,FO IT2=.0001,FO IT3=.0001
FO IT4=.0001,FO IT5=.0001
INITIAL TIME=27.9
PRINT CONTROL=4
SS ITERATIONS=60
ALL STATES
INT CONTROLS,R  SG=0,ROLSG=0,YAWSG=0,ALTSG=0
X  SG=0,Y  SG=0
Q  SG=0,PITSG=0
LINEAR ANALYSIS
```

Figure 146 EASY XC-8A ACLS Analysis File For The Landing Simulation (Continued)

```
STEADY STATE
XIC-X
LINEAR ANALYSIS
TINC=.1,TMAX=31.9,INT MODE=6
ALL STATES
INT CONTROLS
FO IT1=0,FO IT2=0,FO IT3=0
FO IT4=0,FO IT5=0
LINEAR ANALYSIS
PRINTER PLOTS,PLOT ON
DISPLAY1
PCUSH,VS,TIME
PTRUNK,VS,TIME
VC TS,VS,TIME
VT TS.VS,TIME
DISPLAY2
ROLSG,VS,TIME
PITSG,VS,TIME
YAWSG,VS,TIME
Q  SG,VS,TIME
DISPLAY3
X  SG,VS,TIME
Y  SG,VS,TIME
ALTSG,VS,TIME
ACELCG,VS,TIME
VKNOTS,VS,TIME
DISPLAY4
ELEOL,VS,TIME
TITLE     XC-8A LANDING SIMULATION
PLOT ID      PAUL R. PERKINS  MS 47-03
SIMULATE
XIC-X
LINEAR ANALYSIS
```

Figure 146 EASY XC-8A ACLS Analysis Tile For The Landing Simulation (Concluded)

side (See Figure 147). The values input into tables ABLTS, XYZTS, DM TS, and IALTS are tabularized by element number in Table 12. These data were gleaned from References 10, 11, 12, and 13. The load elongation curves representing elastic trunk sets I, II, and III for the meridian and hoop direction were derived from Reference 14 and input in Tables SPHTS and STHTS. These tables are reproduced in Figures 148 and 149. The Poisson's ratio for trunk sets I, II, and III were repectively .5, .5 and .3. Data for the pillow brake elements are input in table PM TS. The coefficient of friction for an extended pillow brake element is .8 compared with .097 for a conventional trunk element. The discharge coefficients for the trunk perforations is .62 (CD1TS), for the cushion gap is 1.0 (CDGTS) and the flattened trunk is .2 (CD2TS). The source for the first two discharge coefficients was Reference 8 pg. 130 and the source for the flattened trunk discharge coefficient was the EASY Jindivik model.

The XC-8A ACS was powered by two ASP-10 fans. The fan map and stall line for the ASP-10 fan are input in Tables FANFN and STAFN respectively. Plots of these data are shown in Figures 150 and 151. The basic ASP-10 fan map was obtained from Reference 15. These curves found in Reference 15 were extrapolated for pressure ratios below the stall line inorder to cover the dynamic range expected during landing. Whenever the fan is operating in this extrapolated region the component FN will print at each print interval "THE FAN IS STALLED" to notify the user.

A STEADY STATE command was performed to initialize the model prior to requesting SIMULATE command. ALTSG, Q SG, R SG, ROLSG PITSG, YAWSG, X SG AND Y SG states were frozen during the steady state analysis. Even though the ALTSG and Q SG states were frozen the pseudo-intergral controllers, IT1 and IT2, varied their respective rates. This was accomplished by implementing into the error signals (FO IT1 and FO IT2) the Fortran coding for the rates of ALTSG and Q SG. The outputs of the pseudo

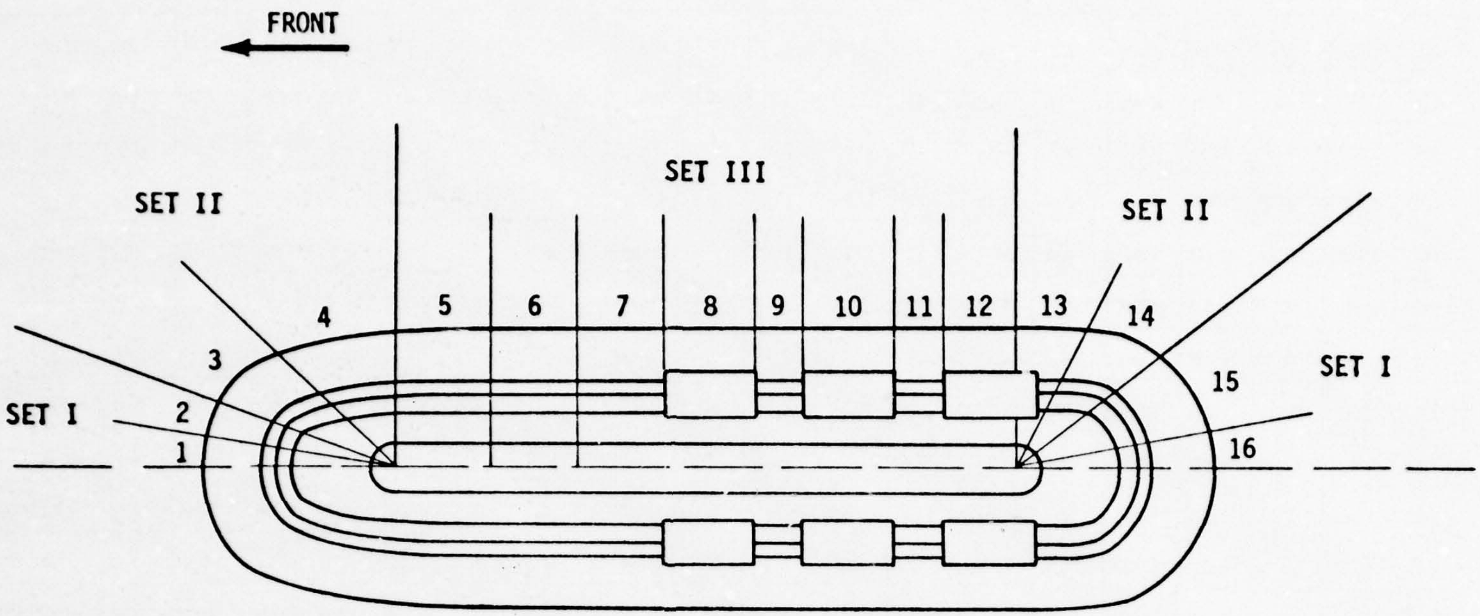


Figure 147 The XC-8A ACS Trunk, Divided Into Sets And Elements

TABLE 12 XC-8A ACS TRUNK DATA

XC-8A LANDING SIMULATION   c.g. = 28.0% $\bar{c}$ FSCG = 343.41

	LENGTHS:			DISTANCE OF A FROM c.g.				ELEMENT WIDTH	UNSTRETCHED VALUES			INITIAL STRAINS		LOCATIONS OF: A			LOCATIONS OF: B			
Set-Element	A (in)	B (in)	L (in)	XA (in)	YA (in)	ZA (in)	BET (DEG)	D (IN OR DEG)	AP	LP (in)	LH (in)	ETI (HOOP)	EPI (MERIDIAN)	WL	BL	FS	WL	BL	FS	MU
I-1	71.9	0.0	71.9	148.91	1.6	62.5	80.0	20.0	.02189	21.0	23.0	0.0	.05	97.5	1.6	194.5	97.5	14.0	123.7	.097
I-2	71.9	0.0	71.9	147.81	4.5	62.5	60.0	20.0	.02189	21.0	23.0	0.0	.05	97.5	4.5	195.6	97.5	40.45	133.3	.097
II-3	47.1	10.0	57.1	145.81	6.9	62.5	40.0	20.0	.01911	22.5	14.0	0.0	.12	97.5	6.9	197.6	107.5	43.0	167.3	.097
II-4	47.1	10.0	57.1	142.41	8.7	62.5	15.0	30.0	.01911	22.5	14.0	0.0	.12	97.5	8.7	201.0	107.5	48.6	188.8	.097
III-5	44.2	15.0	56.8	124.56	9.0	62.5	0	31	.01911	22.25	14.0	0.0	.17	97.5	9.0	218.85	112.5	53.2	218.85	.097
III-6	44.2	15.0	56.8	93.56	9.0	62.5	0	31	.01911	22.25	14.0	0.0	.17	97.5	9.0	249.85	112.5	53.2	249.85	.097
III-7	44.2	15.0	56.8	62.56	9.0	62.5	0	31	.01911	22.25	14.0	0.0	.17	97.5	9.0	280.85	112.5	53.2	280.85	.097
III-8	44.2	15.0	56.8	29.41	9.0	62.5	0	34.0	.0025	19.5	21.0	0.0	.17	97.5	9.0	314.0	112.5	53.2	314.0	.097
III-9	44.2	15.0	56.8	5.91	9.0	62.5	0	13.25	.01911	23.5	13.0	0.0	.17	97.5	9.0	337.5	112.5	53.2	337.5	.097
III-10	44.2	15.0	56.8	-17.84	9.0	62.5	0	34.0	.0025	19.5	21.0	0.0	.17	97.5	9.0	361.25	112.5	53.2	361.25	.097
III-11	44.2	15.0	56.8	-41.34	9.0	62.5	0	13.25	.01911	23.5	13.0	0.0	.17	97.5	9.0	384.75	112.5	53.2	384.75	.097
III-12	44.2	15.0	56.8	-65.09	9.0	62.5	0	34.0	.0025	19.5	21.0	0.0	.17	97.5	9.0	408.5	112.5	53.2	408.5	.097
II-13	47-1	10.0	57.1	-83.79	8.8	62.5	-12.5	22.5	.01911	22.5	14.0	0.0	.12	97.5	8.8	427.2	107.5	54.7	437.4	.097
II-14	47.1	10.0	57.1	-87.09	7.4	62.5	-35.0	22.5	.01911	22.5	14.0	0.0	.12	97.5	7.4	430.5	107.5	46.0	457.5	.097
I-15	71.9	0.0	71.9	-89.49	4.8	62.5	-57.5	25.0	.02189	21.0	23.0	0.0	.05	97.5	4.8	432.9	97.5	43.5	493.5	.097
I-16	71.9	0.0	71.9	-90.79	1.6	62.5	-80.0	20.0	.02189	21.0	23.0	0.0	.05	97.5	1.6	434.2	97.5	14.0	505.0	.097

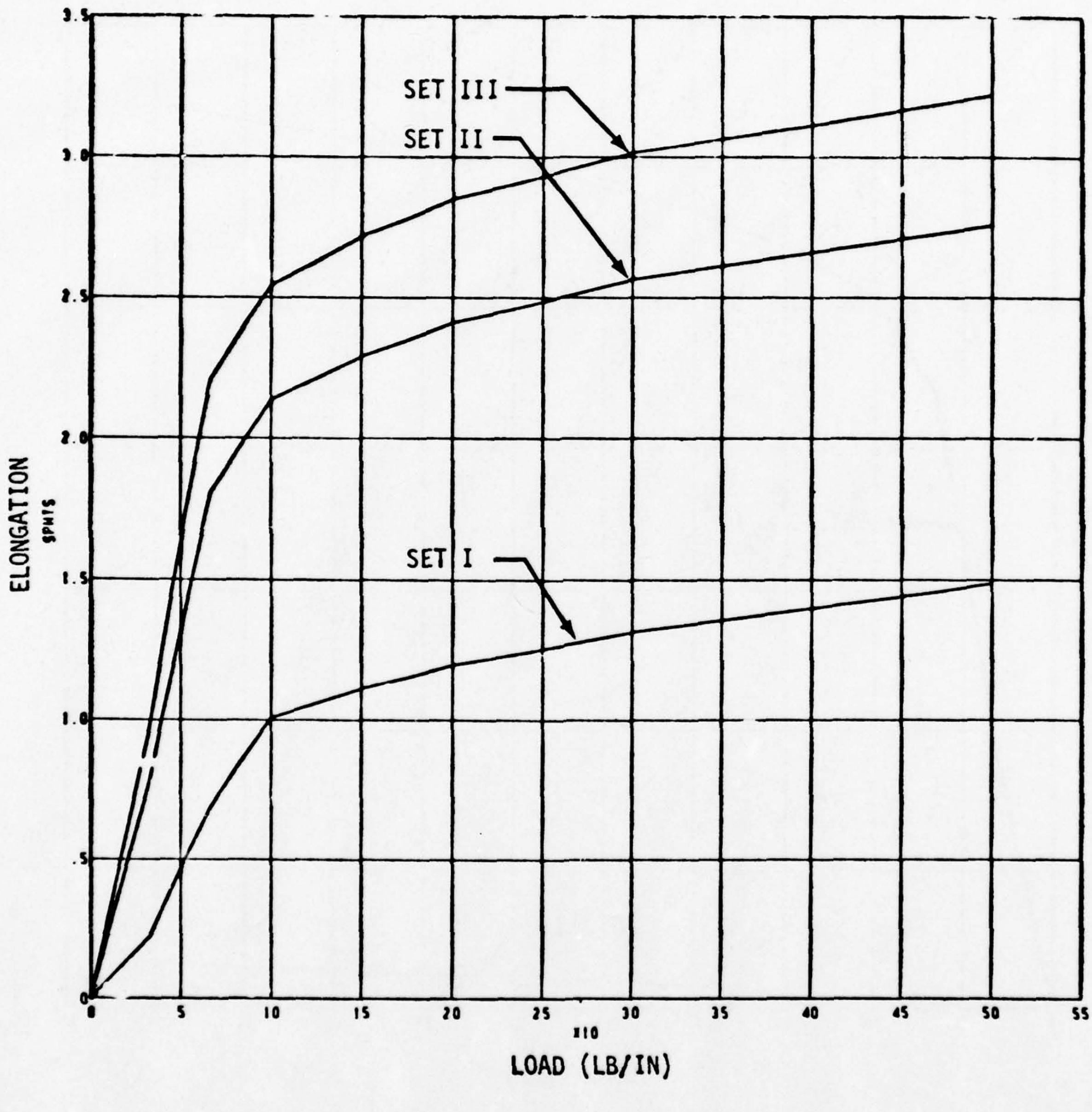


Figure 148 Table SPHTS Load Elongation Curves in the Meridian Direction for Trunk Element Sets I, II and III

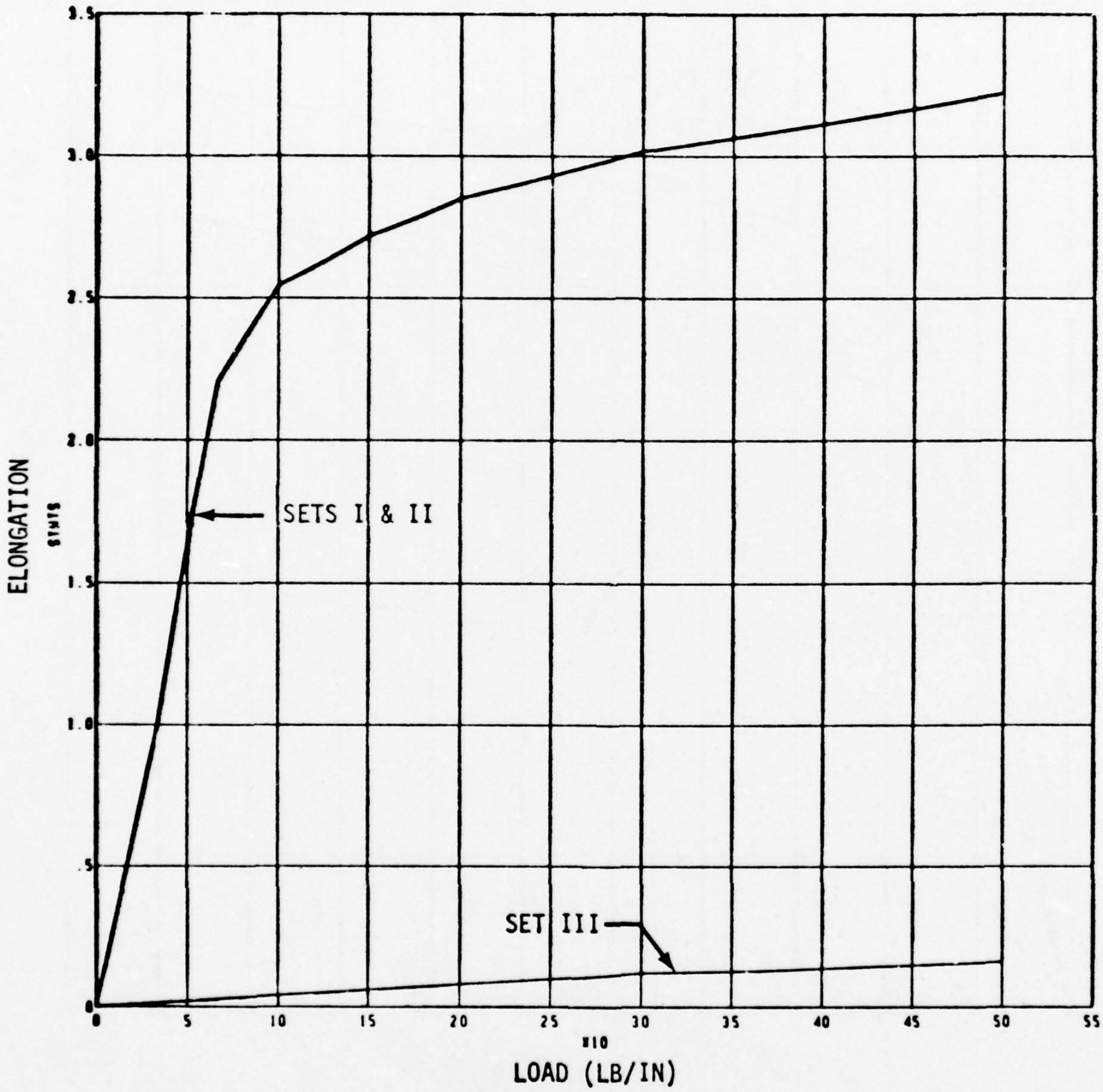


Figure 149 Table STHTS Load Elongation Curves in the Hoop Direction for the Trunk Element Sets I, II and III

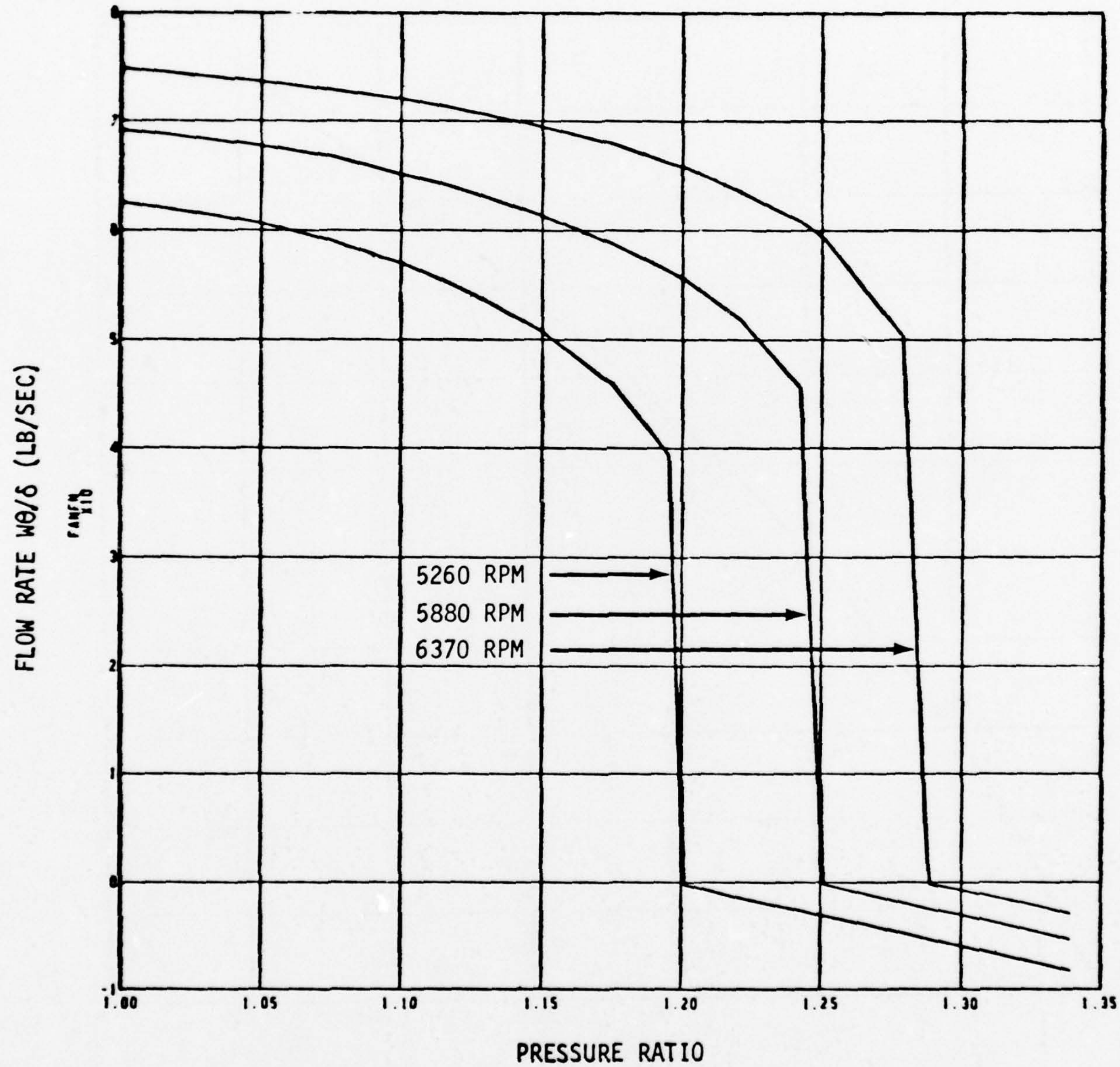


Figure 150 Table FANFN ASP-10 Fan Map

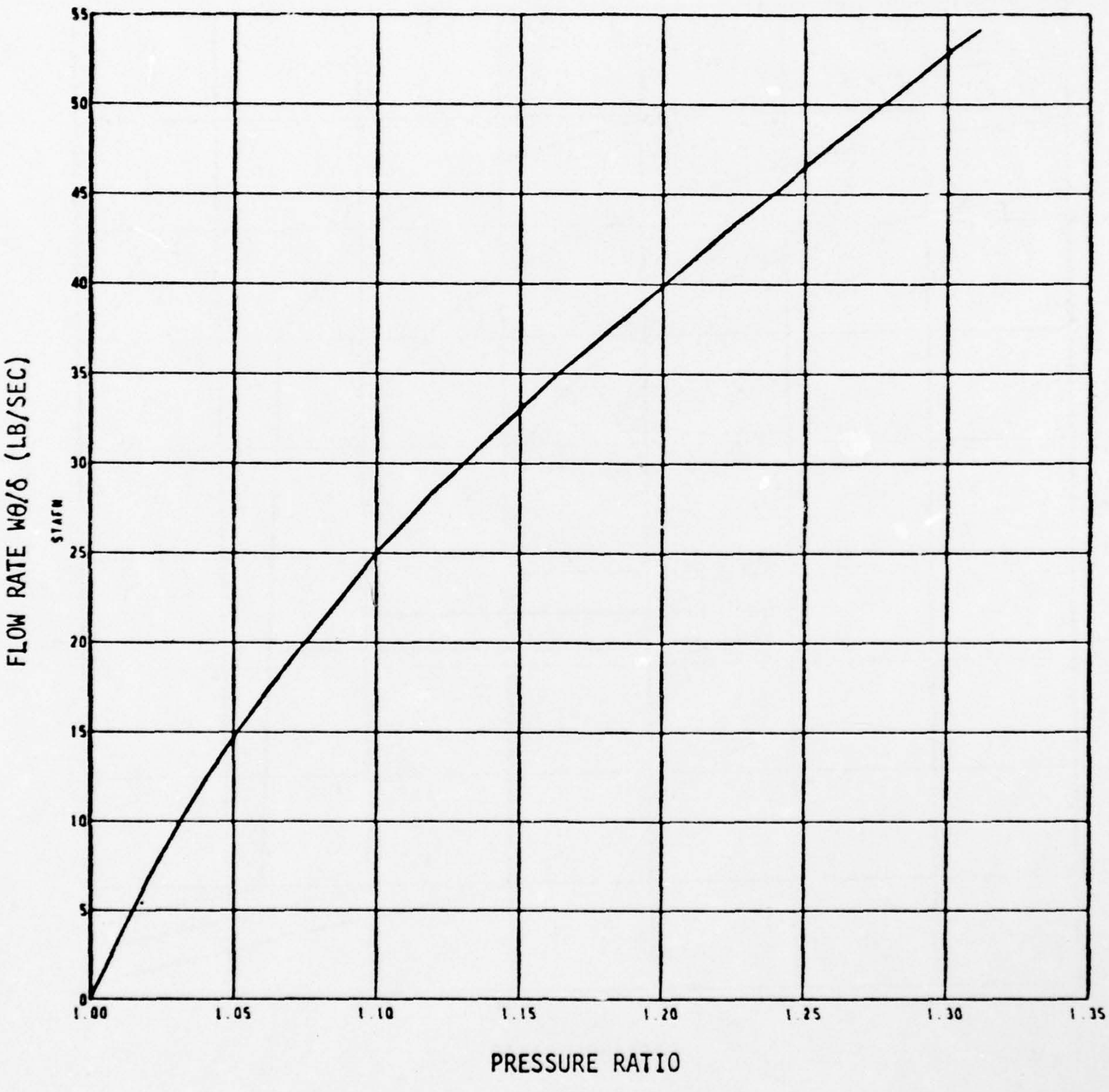


Figure 151 Table STAFN ASP-10 Stall Curve

integral controllers set the engine thrust, elevator deflection, rudder deflection, aileron deflection and cushion trim valve effective area. Their values are listed in Table 11.

9.3 Simulation Description and Results

The EASY XC-8A ACLS model was initialized for landing with the values listed in Table 11. Only the pseudo-integral controllers were frozen during the landing simulation. The model simulation results prior to touchdown show good correlation when compared with flight test data. This indicates proper model initialization for the landing simulation.

Figure 152 a, b, c, d, and e compares the XC-8A aircraft flight test data with the EASY simulation results for aircraft pitch, Q angular rate, vertical acceleration of the c.g., trunk and cushion pressures and elevator deflection. At 28.3 sec the aircraft pitch and Q shown in Figures 152-a and -b demonstrate a divergence between the aircraft flight test data and the EASY simulation. Since the integral of Q is the aircraft pitch angle and the derivative of Q is proportional to the pitching moment, it is apparent that the test aircraft experienced a large negative pitching moment which drove Q negative thus decreasing the pitch from 3.2 degrees to .35 degrees during landing. The EASY simulation on the other hand experienced a positive pitching moment during landing which made Q positive and thus increased the aircraft pitch attitude. Since the initial trunk and cushion pressure spikes in Figure 152-d occur at the same time and have approximately the same magnitudes, it can be concluded that the EASY model accurately simulates the XC-8A pneumatic system and is therefore not the source of the divergence.

Due to limitations and inadequacies in the XC-8A flight test data it was not possible to resolve the differences with the EASY predictions as noted.

To troubleshoot and resolve differences found in a comparison of test data and computer predictions, highly reliable flight test data are required with

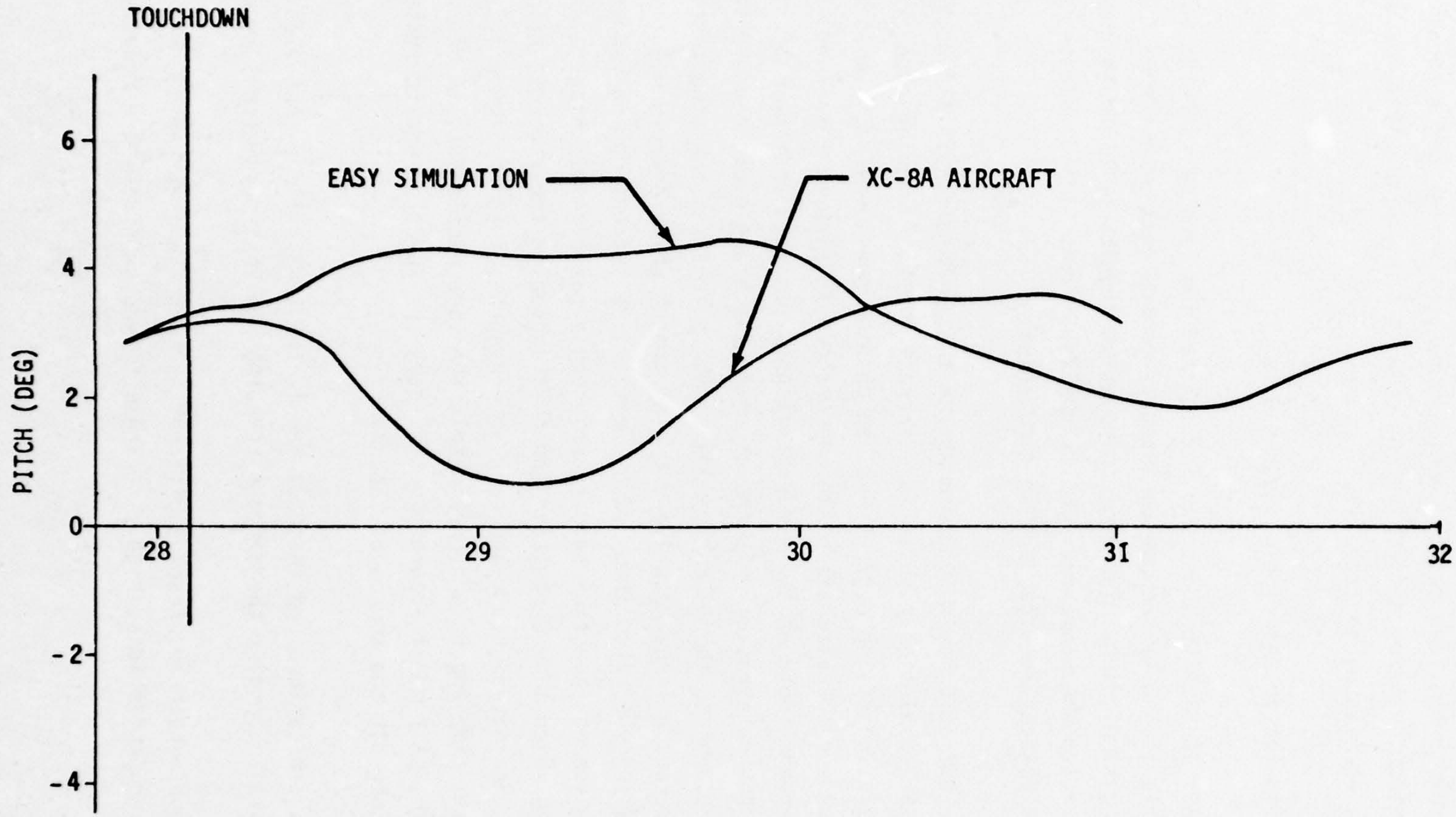


Figure 152(a) XC-8A Landing Simulation A Comparison Of The XC-8A Aircraft With The EASY Simulation Results

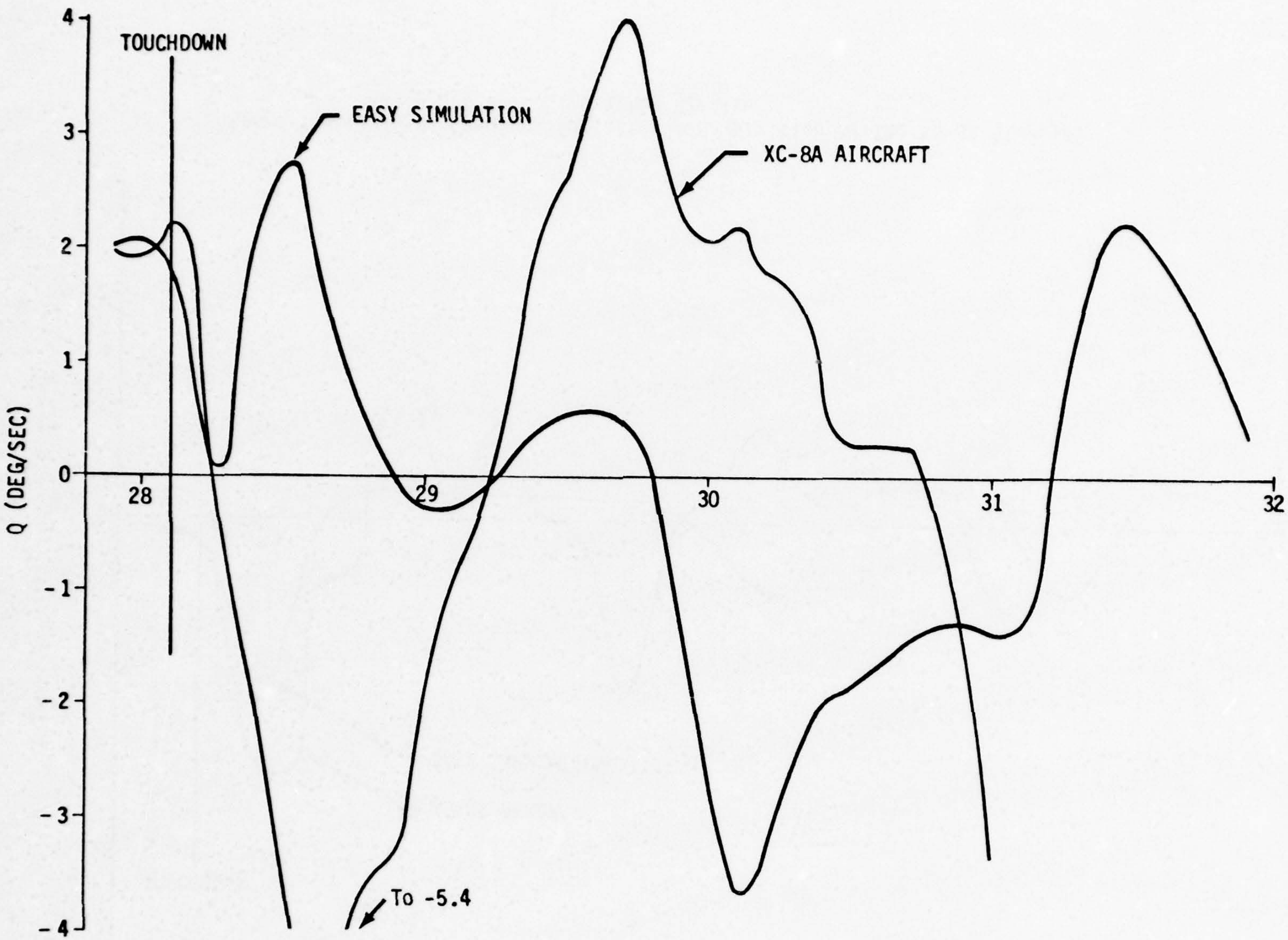


Figure 152 (b) XC-8A Landing Simulation A Comparision Of The XC-8A Aircraft With The EASY Simulation Results

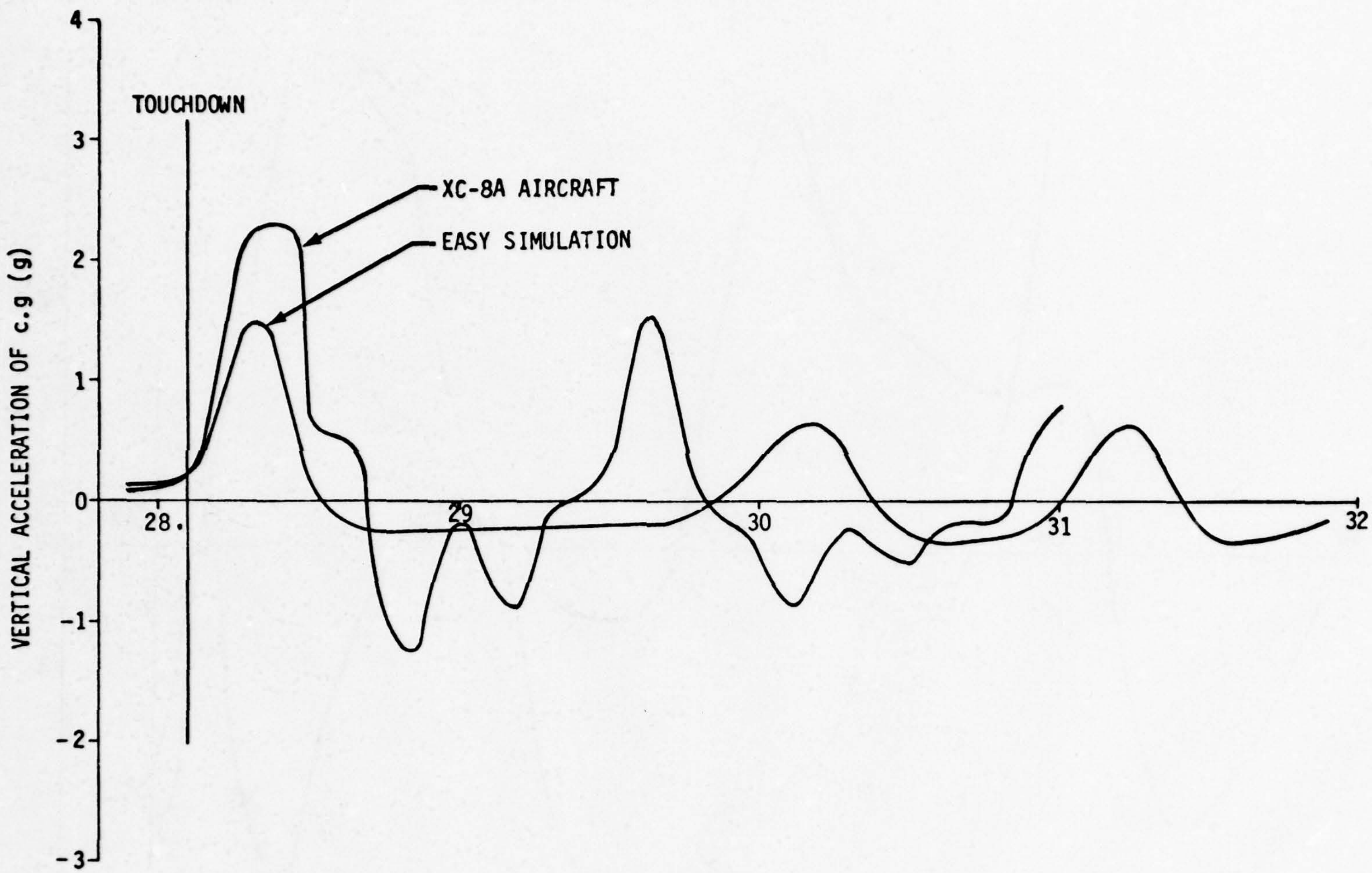


Figure 152(c) XC-8A Landing Simulation A Comparision Of The XC-8A Aircraft With The EASY Simulation Results

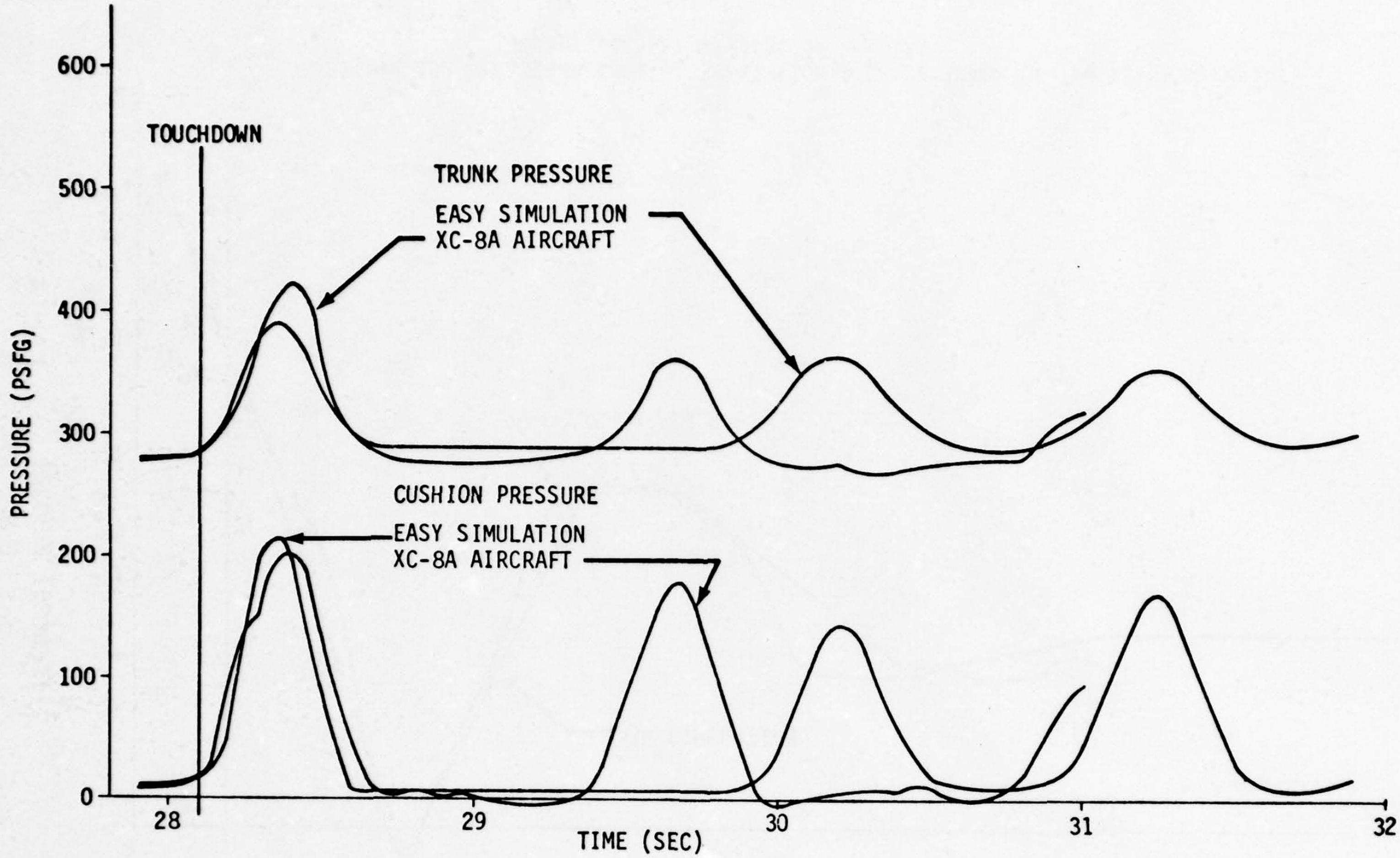


Figure 152(d) XC-8A Landing Simulation A Comparision Of The XC-8A Aircraft With The EASY Simulation Results

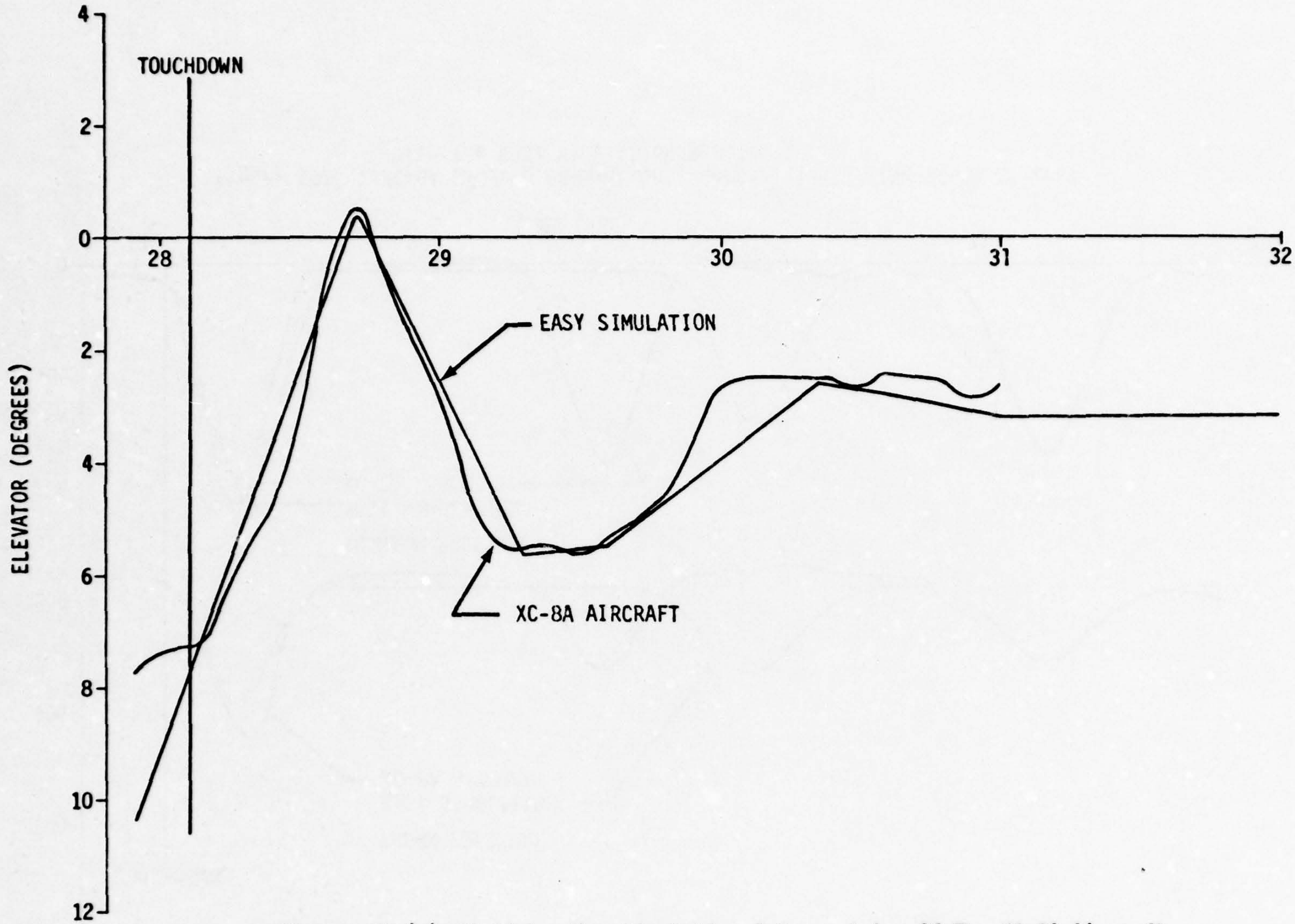


Figure 152 (e) XC-8A Landing Simulation A Comparision Of The XC-8A Aircraft With The EASY Simulation Results

sufficient information of key aircraft system parameters. The XC-8A flight test data did not meet these requirements. The following are the major deficiencies found in the data:

1. Spoiler deployment during touchdown was not recorded. The manner in which aircraft vertical kinetic energy is absorbed or stored during touchdown is affected by the spoiler deployment schedule. A variation in the schedule, especially at high sink rates, will significantly alter the landing characteristics of the aircraft.

2. Engine thrust was not available nor could it be adequately deduced from the available information.

3. Aircraft altitude was recorded from the flight deck altimeter. However, resolution and accuracy were not sufficient to interpret aircraft vertical oscillations during landing.

4. The aircraft angle-of-attack data were considered unreliable. The angle-of-attack vain for the test aircraft was located on a wing tip boom. This boom is subjected to considerable vertical and pitching motion during landing due to wing flexibility.

5. Sink speed could not be directly deduced from the flight test data since aircraft angle-of-attack data could not be used. Sink speed is calculated from the true air speed, the aircraft pitch angle and the aircraft angle-of-attack.

6. Ducted air flow from the ASP-10 was not considered reliable since the recorded flow in the flight test data (126%) was significantly higher then the capability of the unit.

9.4 Conclusions and Recommendations

1. Differences exist in the comparison of XC-8A flight test data and EASY simulated touchdown time histories. However, the flight test data are insufficient to resolve these differences.

2. Due to the lack of adequate flight test data, verification of the EASY XC-8A touchdown simulation was not possible. The simulation, based on a qualitative comparison with flight test data, appears to be functioning properly. The pneumatic system (trunk and cushion pressure) correlates well with flight test data.

3. It is recommended that future efforts to verify the EASY/ACLS models use test data events with sufficient instrumentation to extract and record data necessary to troubleshoot differences that may exist between test data and model predictions.

   Scale model ACLS tests such as those planned using the AFFDL/FEMB rotating arm test facility are considered ideal for the purpose of verifying the EASY models.

SECTION X

XC-8A TAKEOFF SIMULATION

10.1 Objectives

The takeoff simulation objective was to simulate a typical XC-8A takeoff by modifying the EASY XC-8A ACLS model created for the landing simulation. The EASY ACLS takeoff simulation was compared with actual flight test data to validate the computer model.

10.2 Technical Approach

A takeoff with 30 degree flaps from the XC-8A flight #F-033175-1 was selected for simulation. A complete listing of the takeoff configuration (corresponding to time=101556.0 in the flight test data) are listed in Table 13.

10.2.1 Model File

Ten standard EASY components and several pages of Fortran coding were assembled to form the EASY XC-8A ACLS takeoff simulation model. The EASY XC-8A model file and computer generated model schematic for the takeoff simulation are given in Figures 153 and 154. The aerodynamics and ACLS components remain unchanged from the landing simulation model file (see 9.2.1). There is only one pseudo-intergral controller (see Appendix B) in the takeoff model. This controller varies the trunk cushion trim valve area (CAVTS) to maintain the trunk pressure at 16.928 psia. Component TB was added to input thrust and elevator schedules for the simulation.

TABLE 13 XC-8A TAKEOFF CONFIGURATION

(TIME=10.0 SEC)

FLAPS	30 DEG
TRUE AIRSPEED	27.37 FT/SEC
ALTITUDE (c.g.)	8.0 FT
GROSS WEIGHT	35841.8 LBS
CENTER OF GRAVITY (c.g.)	28.1% MAC
FSCG	343.53 IN
WLCG	160.0 IN
MOMENTS OF INERTIA:	
IXX	183700 SLUGS-FT^2
IYY	228200 SLUGS-FT^2
IZZ	365100 SLUGS-FT^2
IXZ	29220 SLUGS-FT^2
THRUST:	
THRRT	5000 LBS
THRLT	5000 LBS
ATTITUDE:	
ANGLE OF ATTACK (α)	1.0 DEG
SIDESLIP ANGLE (β)	0.0 DEG
FLIGHT PATH ANGLE (γ)	0.0 DEG
ROLL (ϕ)	0.0 DEG
PITCH (θ)	1.0 DEG
YAW (ψ)	0.0 DEG
LONGITUDINAL CONTROL-SURFACE DEFLECTIONS:	
ELEVATOR	-25.0 DEG
SPOILER	0.0 DEG
LATERAL CONTROL SURFACE DEFLECTIONS:	
RUDDER	0.0 DEG
AILERON	0.0 DEG

TABLE 13 XC-8A TAKEOFF CONFIGURATION (CONCLUDED)

(TIME=10.0 SEC)

AIR CUSHION PARAMETERS:	
TRUNK PRESSURE	322.8 PSFG
CUSHION PRESSURE	119.3 PSFG
FAN RPM	6186.0 RPM
EFFECTIVE AREA FOR TRUNK TO CUSHION TRIM VALVE	182.7 IN^2

```
MODEL DESCRIPTION    XC-8A TAKEOFF SIMULATION (TASK 5)
FORTRAN STATEMENTS
      COMMON /BMADTS/IERR
      REAL L1,M1,N1
      DATA IERR/0/
ADD PARAMETERS=THETA
FSENG,BLENG,WLENG,FSCG,BLCG,WLCG
FSSKID,BLSKID,WLSKID,SPLOW,SPHIGH,DEFMX,SKIDMU,SKID
TRUNK,CGRAV,FLAPS,SPOIL
ROOTC,ASPECT,SH,FS25CR,WL25CR
FS75CR,WL75CR,FSTAIL,WLTAIL
ADD VARIABLES=CTRT,CTLT,CTTOT,FFBETA
THRRT,THRLT
PCUSH,PTRUNK
COMP,ZFORCE,XFORCE
DELCLGR,DELCDLG,DELCMG
ACELCG,VKNOTS
ADD TABLES
TBCL0,46,TBCL20,46,TBCL40,46
TBCLH,38,TBX,-25,TBLL0,54,TBDEL,-13
LOCATION=58    VA    INPUTS=SG
LOCATION=72    TB
FORTRAN STATEMENTS
      FINIT1=16.928-PT TS
      THRRT=A2 TB
      THRLT=THRRT
      ELEOL=B2 TB
LOCATION=101    IT1
FORTRAN STATEMENTS
      CAVTS=F0 IT1
C*********VKNOTS=THE TRUE AIR SPEED. (KNOTS)
      VKNOTS=VT VA/1.6889
C*********************************************************************
C*********************************************************************
C  THESE FORTRAN STATEMENTS UPDATE THE COS(ALPHA) AND SIN(ALPHA)
C  WITHOUT CHANGING THE VALUE OF ALSVA. THIS PROVIDES FOR CORRECT
C  RESOLUTION OF THE LIFT AND DRAG FORCES FROM THE STABILITY
C  AXES INTO THE BODY AXES. THESE FIVE FORTRAN STATEMENTS
C  MUST BE IMPLEMENTED WHENEVER THE AERODYNAMIC
C  COEFFICIENTS OR DERIVATIVES ARE INPUT AS FUNCTIONS
C  OF THE ANGLE OF ATTACK.
C  CAUTION- THESE FORTRAN STATEMENTS CANNOT BE USED WHEN THE
C  *******  AERODYNAMIC DERIVATIVES ARE REFERENCED TO THE BODY AXIS.
      RPD=.01745329
      SALVA=SIN(AL VA*RPD)
      CALVA=COS(AL VA*RPD)
      PO VA=P  SG*CALVA+R  SG*SALVA
      RO VA=R  SG*CALVA-P  SG*SALVA
C*********************************************************************
```

Figure 153 EASY XC-8A ACLS Model File For The Takeoff Simulation

```
C***********************************************************************
C THE FOLLOWING IF STATEMENTS LIMIT THE ALLOWABLE DEFLECTIONS
C OF THE ELEVATOR,RUDDER,FLAPS AND AILERION WHEEL. THEY ALSO
C LIMIT THE ENGINE THRUST TO A SPECIFIED RANGE.
C*********RANGE OF ELEVATOR DEFLECTION= -25.0 TO +15.0 DEG.
C          (A POSITIVE ELEVATOR DEFLECTION IS DOWN)
      IF(ELEOL.LT.-25.0) ELEOL=-25.0
      IF(ELEOL.GT.15.0) ELEOL=15.0
C*********RANGE OF RUDDER DEFLECTION= -20.0 TO +20.0 DEG.
C          (A POSITIVE RUDDER DEFLECTION IS TO THE PILOTS LEFT)
      IF(RUDDL.LT.-20.0) RUDDL=-20.0
      IF(RUDDL.GT.20.0) RUDDL=20.0
C*********RANGE OF FLAP DEFLECTION= 0.0 TO +40.0 DEG.
      IF(FLAPS.LT.0.0) FLAPS=0.0
      IF(FLAPS.GT.40.0) FLAPS=40.0
C*********RANGE OF AILERION WHEEL ROTATION= -75.0 TO +75.0 DEG.
C          (A POSITIVE AILERION DEFLECTION TIPS THE RIGHT WING DOWN)
      IF(AILDL.LT.-75.0) AILDL=-75.0
      IF(AILDL.GT.75.0) AILDL=75.0
C*********THE MAXIMIUM THRUST PER ENGINE=+9000.0 LBS..
      IF(THRRT.GT.9000.0) THRRT=9000.0
      IF(THRLT.GT.9000.0) THRLT=9000.0
C*********THE MAXIMIUM REVERSE THRUST PER ENGINE=-9000.0
      IF(THRRT.LT.-9000.0) THRRT=-9000.0
      IF(THRLT.LT.-9000.0) THRLT=-9000.0
C***********************************************************************
C***********************************************************************
C  THESE FORTRAN STATEMENTS CALCULATE THE BODY AXIS FORCES
C  AND MOMENTS PRODUCED BY THE ENGINE THRUST. THE THRUST
C  LOCATION AND ANGLE OF THRUST VECTOR MUST BE SPECIFIED.
C  THE MODEL CAN ACCEPT REVERSE THRUST AND/OR DIFFERENTIAL
C  THRUST SCHEDULES.
C*********XENG,YENG AND ZENG=THE XYZ BODY AXIS MOMENT ARM
C         VECTORS (FEET)
C  THE REQUIRED INPUT PARAMETERS ARE
C*********THRLT=LEFT ENGINE THRUST(LBS.)
C*********THRRT=RIGHT ENGINE THRUST(LBS.)
C*********THETA=THE ANGLE OF THRUST APPLICATION IN THE XZ PLANE
C           (DEGREES)
C*********FSENG,BLENG,WLENG. THESE ARE THE THE FUSELAGE
C         STATION,BUTT LINE, AND WATER LINE LOCATIONS
C         OF THE RIGHT ENGINE THRUST VECTOR  (IN.)
C*********FSCG,BLCG,WLCG. THESE ARE THE FUSELAGE STATION,
C         BUTT LINE AND WATER LINE LOCATIONS OF THE C.G..(IN.)
      RPD=.01745329
      FLPRAD=FLAPS*RPD
      THRAD=THETA*RPD
      COSINE=COS(THRAD)
      SINE=SIN(THRAD)
```

Figure 153 EASY XC-8A ACLS Model File For The Takeoff Simulation (Continued)

```
      XENG=(FSCG-FSENG)/12.0
      YENG=(BLENG-BLCG)/12.0
      ZENG=(WLCG-WLENG)/12.0
      THDIF=THRLT-THRRT
      THRT=THRRT
      THLT=THRLT
      IF(THRT.LT.0.0) THRT=0.0
      IF(THLT.LT.0.0) THLT=0.0
      DIF=THLT-THRT
      FX1=(THRLT+THRRT)*COSINE
      FY1=(.002+.0183*FLPRAD)*DIF
      FZ1=-(THRLT+THRRT)*SINE
      T1=-.0176*(THRLT+THRRT)
      T2=(.0125+.1095*FLPRAD)*DIF
      TX1=T1+T2
      TY1=FX1*ZENG-FZ1*XENG
      TZ1=(.1575-.0322*FLPRAD)*THDIF
C*********************************************************************
C*********************************************************************
C THESE FORTRAN STATEMENTS DETERMINE THE BODY AXIS FORCES AND
C TORQUES PRODUCED WHEN A WING SKID TOUCHES THE GROUND.THE REQUIRED
C INPUT PARAMETERS ARE
C*********FSSKID,BLSKID,WLSKID ARE THE FUSELAGE STATION,BUTT LINE
C         AND WATER LINE LOCATIONS OF THE TIP OF THE UNDEFORMED
C         RIGHT SKID. IT IS ASSUMED THAT THE SKIDS ARE AT SYMMETRIC
C         LOCATIONS W.R.T. THE X-Z PLANE. (IN.)
C*********FSCG,BLCG,WLCG. THESE ARE THE FUSELAGE STATION,
C         BUTT LINE AND WATER LINE LOCATIONS OF THE C.G.. (IN.)
C*********SPLOW=LOW SPRING CONSTANT OF THE SKID (I.E. NORMAL SPRING
C         RATE).(LBS/FT)
C*********SPHIGH=HIGH SPRING CONSTANT WHICH REPRESENTS THE SPRING
C         CONSTANT OF THE BOTTOMED OUT SKID.(LBS/FT)
C*********DEFMX=THE MAXIMIUM NORMAL DEFLECTION OF THE SKID.(FT)
C         DEFLECTIONS BEYOND DEFMX ARE CONTROLLED BY THE
C         HIGHER SPRING RATE.
C*********SKIDMU=THE COEFFICIENT OF DYNAMIC FRICTION BETWEEN
C         THE SKID AND THE GROUND.
C*********SKID-THE SKID IS ELIMINATED FROM THE CALCULATIONS
C         WHEN SKID=0.0.
      FX2=0.0
      FY2=0.0
      FZ2=0.0
      TX2=0.0
      TY2=0.0
      TZ2=0.0
      COMP=0.0
      XFORCE=0.0
      ZFORCE=0.0
      IF(SKID.EQ.0.0) GO TO 11
```

Figure 153 EASY XC-8A ACLS Model File For The Takeoff Simulation (Continued)

```
      XSKID=(FSCG-FSSKID)/12.0
      YSKID=(BLSKID-BLCG)/12.0
      ZSKID=(WLCG-WLSKID)/12.0
      IF(ROLSG.LT.0.0) YSKID=-YSKID
      RPD=.01745329
      CP=COS(PITSG*RPD)
      SP=SIN(PITSG*RPD)
      CR=COS(ROLSG*RPD)
      SR=SIN(ROLSG*RPD)
      CY=COS(YAWSG*RPD)
      SY=SIN(YAWSG*RPD)
C*********HGT IS THE HEIGHT OF THE SKID ABOVE THE GROUND. IF HGT
C         IS NEGATIVE THE SKID SPRING IS COMPRESSED.
      HGT=+XSKID*SP-YSKID*SR*CP-ZSKID*CR*CP+ALTSG
      IF(HGT.GE.0.0) GO TO 11
      COMP=-HGT/(SQRT(1.0-(SR*CP)**2.))
C*********COMP IS THE AMOUNT THAT THE SKID SPRING IS COMPRESSED.
C         COMP DOES NOT EQUAL HGT BECAUSE THE AIRCRAFT IS ROTATED
C         W.R.T. THE EARTH. THE DENOMINATOR OF THIS ASSIGNMENT
C         STATEMENT IS THE COSINE OF THE ANGLE WHICH THE SKID
C         MAKES FROM VERTICAL.
      HIGH=SPHIGH
      IF(COMP.LT.DEFMX) HIGH=SPLOW
C         THE EARTH AXIS FORCES PRODUCED BY THE COMPRESSION OF
C         THE SKID SPRING ARE DETERMINED BELOW.
      ZFORCE=-(DEFMX*SPLOW+(COMP-DEFMX)*HIGH)
      XFORCE=SKIDMU*ZFORCE
      IF(IERR.NE.1) GO TO 211
      WRITE(6,210)
 210  FORMAT(10X,35HA WING SKID IS TOUCHING THE GROUND.)
 211  CONTINUE
C         THE BODY AXIS FORCES ARE CALCULATED BELOW.
      FX2=XFORCE*CP*CY-ZFORCE*SP
      FY2=XFORCE*(SR*SP*CY-CR*SY)+ZFORCE*SR*CP
      FZ2=XFORCE*(CR*SP*CY+SR*SY)+ZFORCE*CR*CP
C         THE BODY AXIS TORQUES ARE CALCULATED BELOW.
      TX2=YSKID*FZ2-ZSKID*FY2
      TY2=ZSKID*FX2-XSKID*FZ2
      TZ2=XSKID*FY2-YSKID*FX2
  11  CONTINUE
C THE NEXT 6 STATEMENTS SUM THE FORCES AND MOMENTS PRODUCED
C BY THE ENGINES AND SKIDS.
      FX1S2=FX1+FX2
      FY1S2=FY1+FY2
      FZ1S2=FZ1+FZ2
      TX1S2=TX1+TX2
      TY1S2=TY1+TY2
      TZ1S2=TZ1+TZ2
C*********************************************************************
```

Figure 153 EASY XC-8A ACLS Model File For The Takeoff Simulation (Continued)

```
C**************************************************************************
C THESE FORTRAN STATEMENTS DETERMINE THE AERODYNAMIC
C DERIVATIVES AND COEFFICIENTS USED IN COMPONENTS OL AND DL.
C THE CALCULATED DERIVATIVES AND COEFFICIENTS ARE ALL WITH
C RESPECT TO THE STABILITY AXIS AND ARE ALL NONDIMENSIONAL
C VALUES. THE MOMENT TERMS ARE FOR A C.G. LOCATION AT 40
C PERCENT MAC. XP1OL CORRECTS THE DATA FOR A DIFFERENT
C C.G. LOCATION.
C THE XC-8A AERODYNAMIC MODEL WAS OBTAINED FROM G.KURYLOWICH
C AIR FORCE FLIGHT DYNAMICS LABORATORY.
C         *********************************************
C         * THE FOLLOWING VARIABLES AND PARAMETERS    *
C         * ARE NEEDED TO DETERMINE THE AERODYNAMIC   *
C         * COEFFICIENTS AND DERIVATIVES. THESE ARE   *
C         * NOT EASY STANDARD COMPONENT VARIABLES OR  *
C         * PARAMETERS.                               *
C         *   TRUNK-  THE TRUNK IS DEFLATED WHEN      *
C         *         TRUNK=0.0, AND FULLY INFLATED     *
C         *         WHEN TRUNK=1.0.                   *
C         *   CGRAV-  THE LOCATION OF THE C.G.. IT    *
C         *         IS  INPUT AS A PERCENT MAC.       *
C         *   FLAPS-  THE ANGULAR DEFLECTION OF THE   *
C         *         FLAPS,INPUT IN DEGREES.           *
C         *   SPOIL-  THE FRACTION OF THE TOTAL       *
C         *         SPOILER DEFLECTION. THE RANGE     *
C         *         OF SPOIL IS FROM 0.0 TO 1.0. THE  *
C         *         SPOILERS ARE FULLY DEPLOYED WHEN  *
C         *         SPOIL=1.0, AND FULLY RETRACTED    *
C         *         WHEN SPOIL=0.0.                   *
C         *********************************************
C THIS NEXT GROUPING OF FORTRAN STATEMENTS EVALUATES
C FREQUENTLY USED EXPRESSIONS PRESENT IN THE AERODYNAMIC
C EQUATIONS.
C*********CTRT IS THE RIGHT ENGINE COEFFICIENT OF THRUST
C*********CTLT IS THE LEFT ENGINE COEFFICIENT OF THRUST.
C*********CTTOT IS THE TOTAL COEFFICIENT OF THRUST.
      CTRT=THRRT/QS VA
      CTLT=THRLT/QS VA
      CTTOT=CTRT+CTLT
C*********IF CTTOT IS NEGATIVE,CTTOT IS THEN SET TO ZERO.
      IF(CTTOT.LT.0.0) CTTOT=0.0
      CTT2=CTTOT**2.
      RPD=.01745329
      ALRAD=AL VA*RPD
      BERAD=BE VA*RPD
      COS2BE=(COS(BERAD))**2.
      SIN2BE=(SIN(BERAD))**2.
      BEABS=ABS(BERAD)
      FFBETA=ABS(1.-.955*(BEABS-.523))
```

Figure 153 EASY XC-8A ACLS Model File For The Takeoff Simulation (Continued)

```
      IF(BEABS.LE..523) FFBETA=1.0
      FLPRAD=FLAPS*RPD
      FLP2=FLPRAD**2.
      ABZF=(WLCG-160.)/12.0
      ABZC=(WLCG-74.6)/12.0
      PIH=3.141592654/2.0
C AERODYNAMIC CONTRIBUTIONS FOR THE APUS,DIFFUSERS AND CUSHION.
C ( A 40 PERCENT C.G. LOCATION WAS USED FOR ALL TERMS.)
      ABCL=1.59/QC VA
      IF(ALTSG.LE.11.0) ABCL=0.0
      ABSB=ABS(SIN(BE VA))
      CDMON=5.19/VT VA
C CUSHION RETRACTED INCREMENTS OGE FANS CLOSED
      RETCD=.004
      RETCY=0.0
      RETCM=-.004*(ABZF-7.9*ALRAD)/C  OL
      RETCLL=0.0
      RETCN=0.0
C CUSHION EXTENDED INCREMENTS FAN CLOSED
      EXTCD=.02
      EXTCY=-.05*ABSB
      EXTCM=-.02*(ABZC-5.54*ALRAD)/C  OL
      EXTCLL=-EXTCY*ABZC/B  DL
      EXTCN=EXTCY*5.54/B  DL
C CUSHION EXTENDED INCREMENTS WITH FANS OPERATIONAL
      EMCD=CDMON+ABCL*ALRAD
      EMCY=-CDMON
      EMCM=-CDMON*(ABZF-5.6*ALRAD)/C  OL+ABCL*5.54/C  OL
      EMCLL=-EMCY*ABZF/B  DL
      EMCN=EMCY*5.6/B  DL
C * * * * * * * * * * * * * * * * * * * * * * * * *
C
C AERODYNAMIC LIFT DERIVATIVES AND COEFFICIENTS
C
C********** BASIC LIFT COEFFICIENT **********
      Z1=(.35+2.75*FLPRAD-.616*FLP2+(.15+1.865*FLPRAD
     1-.821*FLP2)*CTTOT+(5.27+.72*CTTOT)*ALRAD)
     2*COS2BE
      ZO OL=-Z1
C********** ELEVATOR DERIVATIVE **********
      ZDEOL=-.566*COS2BE
C********** SPOILER DERIVATIVE **********
      ZSPOL=-(-.25-.573*FLPRAD)
      SPOOL=SPOIL/RPD
C        SPOIL IS DIVIDED BY RPD TO NEGATE THE EFFECTS
C        OF THE AUTOMATIC DEGREE TO RADIAN CONVERSION.
C********** TRUNK COEFFICIENT **********
      ZTROL=-TRUNK*ABCL
C * * * * * * * * * * * * * * * * * * * * * * * * *
```

Figure 153 EASY XC-8A ACLS Model File For The Takeoff Simulation (Continued)

```
C
C AERODYNAMIC DRAG DERIVATIVES AND COEFFICIENTS
C
C********** BASIC DRAG COEFFICIENT **********
      CL2=(ZO OL+ZDEOL*ELEOL*RPD+ZTROL+ZSPOL*SPOOL*RPD)**2.
      X1=(.026+.0394*CL2+(.115+.46*CTTOT-.12*CTT2
     1)*FLPRAD)*COS2BE
      IF(X1.LT.0.0) X1=0.0
C           MODIFICATION OF BASIC DRAG COEFFICIENT
C           WHEN VT VA IS LESS THAN 70.0 FPS.
      X2=0.0
      IF(VT VA.GT.70.0) GO TO 15
      X2=(.229*FLPRAD+.176*FLP2)*CTTOT
      B=1.0-(VT VA-50.0)/20.0
      IF(50.0.LE.VT VA.AND.VT VA.LE.70.0) X2=X2*B
   15 CONTINUE
      XO OL=-X1-X2
C********** TRUNK COEFFICIENT **********
      XTROL=-(RETCD+TRUNK*(EXTCD+EMCD))
C********** SPOILER DERIVATIVE **********
      XSPOL=(-.0125-(.1*FLPRAD))
C * * * * * * * * * * * * * * * * * * * * * * * * *
C
C AERODYNAMIC SIDEFORCE DERIVATIVES AND COEFFICIENTS
C
C********** BETA DERIVATIVE **********
      IF(BEABS.GT..436) GO TO 18
      A=-.8-.515*FLPRAD
      B=-.1-.702*FLPRAD
      C=-.57-1.89*FLPRAD+.82*FLP2
      D=.573-1.64*FLPRAD
      IF(FLPRAD.GE..349) D=-1.145
      Y1=A+B*CTTOT+(C+D*CTTOT)*ALRAD
      GO TO 19
   18 A=.08+.225*FLPRAD
      B=(.025+.322*FLPRAD)*CTTOT-.079*BERAD
      IF(ABS(CTTOT).LT..1) B=B+.079*BERAD
      IF(BERAD-PIH) 500,501,501
  500 Y1=-(1.0/BEABS)*(1.38*SIN2BE+A*(1.0
     1-(BEABS-.436)/1.134)+B)
      GO TO 19
  501 Y1=-(1.0/BEABS)*(1.38*SIN2BE+A*(PIH-BEABS)/
     1   1.134+B)
   19 YB DL=Y1
C********** AILERION DERIVATIVE **********
      YDADL=.0277*(.0473+.0545*FLPRAD)*FFBETA
C********** RUDDER DERIVATIVE AND EFFECTIVENESS **********
      YDRDL=.407
      YBRDL=FFBETA
```

Figure 153 EASY XC-8A ACLS Model File For The Takeoff Simulation (Continued)

```
C********** ROLL RATE DERIVATIVE **********
      YP DL=-.06
C********** YAW RATE DERIVATIVE **********
      YR DL=.704
C********** TRUNK COEFFICIENT **********
      YTRDL=RETCY+TRUNK*(EXTCY+EMCY)
C * * * * * * * * * * * * * * * * * * * * * * * *
C
C AERODYNAMIC PITCHING MOMENT DERIVATIVES AND COEFFICIENTS
C
C********** BASIC PITCHING MOMENT COEFFICIENT **********
      IF(CTTOT.GE.1.0) GO TO 20
      A=.2-.2*CTTOT
      B=.251-.036*CTTOT
      C=-.245*CTTOT
      D=-3.69+3.616*CTTOT
      E=-1.645-5.375*CTTOT
      F=4.71*CTTOT
      GO TO 21
  20  A=.2*(CTTOT-1.0)
      B=.215+1.002*(1.0-CTTOT)
      C=-.49+.245*CTTOT
      D=-.074+1.566*(1.0-CTTOT)
      E=-14.04+7.02*CTTOT
      F=9.42-4.71*CTTOT
  21  M1=A+B*FLPRAD+C*FLP2+(D+E*FLPRAD+F*FLP2)
     1*(ALRAD+.1745)**2.
      MO OL=M1
C********** PITCH RATE DERIVATIVE **********
      MQ OL=-38.4
C********** ALPHA DOT DERIVATIVE **********
      MADOL=-7.-5.*CTTOT
      IF(CTTOT.GE.1.0) MADOL=-12.0
C********** TRUNK COEFFICIENT **********
      MTROL=(RETCM+EXTCM+EMCM)*TRUNK
C********** ELEVATOR DERIVATIVE **********
      MDEOL=-2.43*COS2BE
C********** SPOILER DERIVATIVE
      MSPOL=(.02+.1145*FLPRAD)
C * * * * * * * * * * * * * * * * * * * * * * * *
C
C AERODYNAMIC ROLL MOMENT DERIVATIVES AND COEFFICIENTS
C
C********** BETA DERIVATIVE **********
      IF(BEABS.GT..436) GO TO 31
      L1=(-.164+.043*FLPRAD+(.014+.1575*FLPRAD-.0924*FLP2)
     1*CTTOT+(.172+(.1035-.0186*FLPRAD-.082*FLP2)
     2*CTTOT)*ALRAD)
      GO TO 32
```

Figure 153 EASY XC-8A ACLS Model File For The Takeoff Simulation (Continued)

```
 31  A=.01875*FLPRAD
     B=(.0041+.0326*FLPRAD)*CTTOT
     IF(BEABS-PIH) 510,511,511
510  L1=(1.0/BEABS)*(-.022-.235*SIN2BE*COS2BE+
    1A*(1.0-(BEABS-.436)/1.134)+B)
     GO TO 32
511  L1=(1.0/BEABS)*(-.022-.235*SIN2BE*COS2BE+A
    1   *(PIH-BEABS)/1.134+B)
 32  LB DL=L1
C********** ROLL RATE DERIVATIVE **********
     LP DL=-.53-.1*FLPRAD+(.08-.0572*FLPRAD)*CTTOT
    1+.516*ALRAD
C**********YAW RATE DERIVATIVE **********
     LR DL=.15+.587*FLPRAD+.975*ALRAD
C********** AILERION DERIVATIVE **********
     LDADL=(.0473+.0545*FLPRAD)*FFBETA
C********** RUDDER DERIVATIVE AND EFFECTIVENESS PARAMETER **********
     LDRDL=.184*SIN(.265-ALRAD)
     LBRDL=FFBETA
C********** TRUNK COEFFICIENT **********
     LTRDL=RETCLL+TRUNK*(EXTCLL+EMCLL)
C * * * * * * * * * * * * * * * * * * * * * * * * *
C
C AERODYNAMIC YAW MOMENTS DERIVATIVES AND COEFFICIENTS
C
C********** BETA DERIVATIVE **********
     IF(BEABS.GT..436) GO TO 37
     A=.125+.0329*FLPRAD
     B=-.015-.0415*FLPRAD
     C=.057+.52*FLPRAD-.495*FLP2
     D=-.0573-.181*FLPRAD
     IF(FLPRAD.GE..349) D=-.1204+.1315*(FLPRAD-.349)
     N1=A+B*CTTOT+(C+D*CTTOT)*ALRAD
     GO TO 38
 37  A=.02475+.01435*FLPRAD
     B=(.006+.0236*FLPRAD)*CTTOT
     IF(BEABS-PIH) 520,521,521
520  N1=(1.0/BEABS)*(.106*SIN2BE+A*(1.0
    1-(BEABS-.436)/1.134)-B)
     GO TO 38
521  N1=(1.0/BEABS)*(.106*SIN2BE+A*(PIH-BEABS)/1.134-B)
 38  NB DL=N1
C********** YAW RATE DERIVATIVE **********
     NR DL=-.22-.06*FLPRAD-(.129+.308*FLPRAD)*ALRAD
C********** ROLL RATE DERIVATIVE **********
     NP DL=((-.1+.009*CTTOT)*FLPRAD)/.699+
    1(-.5-.145*CTTOT*(FLPRAD-.699)/.699)*ALRAD
C********** AILERION DERIVATIVE **********
     NDADL=-(.0467+.704*ALRAD)*LDADL
```

Figure 153 EASY XC-8A ACLS Model File For The Takeoff Simulation (Continued)

```
C********** RUDDER DERIVATIVE AND EFFECTIVENESS PARAMETER **********
      NDRDL=-.184*COS(.265-ALRAD)
      NBRDL=FFBETA
C********** TRUNK COEFFICIENT **********
      NTRDL=RETCN+TRUNK*(EXTCN+EMCN)
C XP1OL CORRECTS THE AERODYNAMIC DATA FOR A C.G. LOCATION
C DIFFERENT FROM 40 PERCENT MAC USED IN DERIVING THE DATA.
      XP1OL=C  OL*(.01*CGRAV-.4)
C*********************************************************************
C*********************************************************************
C  THE FOLLOWING FORTRAN STATEMENTS EVALUATE THE GROUND EFFECTS
C  FOR THE AERODYNAMIC MODEL. THE PROCEEDURE USED TO EVALUATE
C  GROUND EFFECTS WAS TAKEN FROM DATCOM SECTION 4.7 METHOD 1.
C  THE REQUIRED INPUT PARAMETERS ARE
C*********ROOTC=ROOT CHORD (CR) LENGTH (FEET)
C*********SH=HORIZONTAL STABILIZER AREA (SQUARE FEET)
C*********ASPECT=ASPECT RATIO
C*********FS25CR,WL25CR=THE FUSELAGE STATION AND WATER
C         LINE LOCATIONS OF .25 ROOT CHORD.
C*********FS75CR,WL75CR=THE FUSELAGE STATION AND WATER
C         LINE LOCATIONS OF THE .75 ROOT CHORD.
C*********FSTAIL,WLTAIL=FUSELAGE STATION AND WATER
C         LINE LOCATIONS OF THE HORIZONTAL STABILIZERS
C         AERODYNAMIC CENTER.
C  THE REQUIRED INPUT TABLES ARE
C*********TBCLO,TBCL20,TBCL40= THE TOTAL AIRPLANES COEFFICIENT
C          OF LIFT AT 0.0,20.0,40.0 DEG. FLAPS RESPECTIVELY.
C         THESE TABLES ARE FUNCTIONS OF ALPHA AND CTTOT.
C*********TBCLH=THE HORIZONTAL STABILIZER COEFFICIENTS OF
C         INPUT AS FUNCTIONS OF HORIZONTAL ALPHA AND ELEOL.
C*********TBX=THE X VARIABLES FROM DATCOM FIGURE 4.7.1-14.
C*********TBLLO=THE L/LO-1. VARIABLE FROM DATCOM FIGURE 4.7.1-15.
C*********TBDEL=THE DELTA DELTA CL VARIABLE FROM DATCOM
C         FIGURE 4.7.1-17 FOR SLOTTED FLAPS.
C  ASSUMPTIONS
C         1. SWEEP ANGLE OF THE QUARTER CHORD LINE
C            OF THE WING = 0.0 DEG..
C         2. THE DYNAMIC PRESSURE OF THE HORIZONTAL
C            STABILIZER EQUALS THE DYNAMIC PRESSURE
C            AT INFINITY.
C         3. THE WINGS ARE STRAIGHT AND PERPENDICULAR
C            TO THE X-Z PLANE. THUS H.75CR=H.75B/2.
C            SEE DATCOM FIGURE 4.7.1-14.
C         4. THE MAIN WINGS AERODYNAMIC CENTER IS ASSUMED
C            TO BE AT .25C..
C  IF THE AIRCRAFTS ALTITUDE IS GREATER THAN TWICE THE
C  WING SPAN THE GROUND EFFECTS ARE SKIPPED.
      DELCLGR=0.0
      DELCDLG=0.0
```

Figure 153 EASY XC-8A ACLS Model File For The Takeoff Simulation (Continued)

```
      DELCMG=0.0
      IF(ALTSG.GT.2.*B  DL) GO TO 777
C   THE NEXT GROUPING OF FORTRAN STATEMENTS EVALUATES
C   FREQUENTLY USED VARIABLES IN THE GROUND EFFECTS
C   CALCULATIONS.
C
      SRATIO=SH/S  VA
      EPSILON=AL VA*(.24-.0015*FLAPS)
      TAILA=AL VA-EPSILON
      X75CR=(FSCG-FS75CR)/12.0
      Z75CR=(WLCG-WL75CR)/12.0
      X25CR=(FSCG-FS25CR)/12.0
      Z25CR=(WLCG-WL25CR)/12.0
      XTAIL=(FSCG-FSTAIL)/12.0
      ZTAIL=(WLCG-WLTAIL)/12.0
      RPD=.01745329
      SP=SIN(PITSG*RPD)
      CR=COS(ROLSG*RPD)
      CP=COS(PITSG*RPD)
      CRCP=CR*CP
      H75CR=X75CR*SP-Z75CR*CRCP+ALTSG
      IF(H75CR.LE.11.25) GO TO 777
      H25CR=X25CR*SP-Z25CR*CRCP+ALTSG
      HTAIL=XTAIL*SP-ZTAIL*CRCP+ALTSG
      H75=2.*H75CR/B  DL
      H25=H25CR/ROOTC
      CRB=ROOTC/B  DL
C   THE NEXT GROUPING OF FORTRAN STATEMENTS EVALUATES
C   THE TOTAL AIRCRAFTS COEFFICIENT OF LIFT BY TABLE
C   LOOK UP ROUTINES INPUT AS FUNCTIONS OF ALPHA AND CTTOT.
C   THERE ARE 3 INPUT TABLES FOR FLAP SETTINGS OF 0.0,20.0
C   AND 40.0 DEGREES. A LINEAR INTERPELATION IS MADE
C   FOR FLAP SETTINGS BETWEEN THESE VALUES.
      I=0
      IF(FLAPS.GE.20.) GO TO 2
      T=FLAPS/20.0
      GO TO 4
  2   T=(FLAPS-20.)/20.
  4   A=AL VA
  5   I=I+1
      IF(FLAPS.GE.20.) GO TO 14
      COEL1=TBLU2(A,CTTOT,TBCL0(7),TBCL0(4),TBCL0(17)
     1,1,1,10,3,10,3)
      COEL2=TBLU2(A,CTTOT,TBCL20(7),TBCL20(4),TBCL20(17)
     1,1,1,10,3,10,3)
      GO TO 150
 14   COEL1=TBLU2(A,CTTOT,TBCL20(7),TBCL20(4),TBCL20(17)
     1,1,1,10,3,10,3)
      COEL2=TBLU2(A,CTTOT,TBCL40(7),TBCL40(4),TBCL40(17)
```

Figure 153 EASY XC-8A ACLS Model File For The Takeoff Simulation (Continued)

```
     1,1,1,10,3,10,3)
  150 CL=(COEL2-COEL1)*T+COEL1
      GO TO (10,200,30,40) I
C  CALCULATION OF THE GROUND EFFECTS UPDATE FOR
C  THE LIFT COEFFICIENT.
  10  CLWBH=CL
      CLH=TBLU2(TAILA,ELEOL,TBCLH(7),TBCLH(4),TBCLH(15)
     1,1,1,8,3,8,3)
C*********CLFWB=IS THE WING-BODY LIFT COEFFICIENT INCLUDING
C         FLAP EFFECTS, OUT OF GROUND EFFECT.
      CLFWB=CLWBH-CLH*SRATIO
      X=TBLU1(H75,TBX(4),TBX(15),1,11)
      IF(X.LT.0.0) X=0.0
      ALUP=AL VA+.5
      ALDN=AL VA-.5
      EUP=ALUP*(.24-.0015*FLAPS)
      EDN=ALDN*(.24-.0015*FLAPS)
      AHUP=ALUP-EUP
      AHDN=ALDN-EDN
      A=ALUP
      GO TO 5
  200 CLUP=CL
      A=ALDN
      GO TO 5
  30  CLDN=CL
      CLHUP=TBLU2(AHUP,ELEOL,TBCLH(7),TBCLH(4),TBCLH(15)
     1,1,1,8,3,8,3)
      CLHDN=TBLU2(AHDN,ELEOL,TBCLH(7),TBCLH(4),TBCLH(15)
     1,1,1,8,3,8,3)
C*********CLAWB=IS THE WING-BODY LIFT CURVE SLOPE,PER DEGREE
C         OUT OF GROUND EFFECT. IT IS EVALUATED BY CALCULATING
C         CLFWB FOR AL VA+.5 AND AL VA-.5 DEGREES. CLAWB IS
C         ASSIGNED THE AVERAGE SLOPE CALCULATED FOR THIS
C         ONE DEGREE CHANGE IN ALPHA.
      CLAWB=CLUP-CLDN-SRATIO*(CLHUP-CLHDN)
      CKCL=CLWBH*9.1196
      OLL=TBLU2(H25,CKCL,TBLLO(7),TBLLO(4),TBLLO(19)
     1,1,1,-12,3,12,3)
      R=SQRT(1.+H75**2.)-H75
      DELDEL=TBLU1(H25,TBDEL(4),TBDEL(9),1,5)
      IF(DELDEL.GT.0.0) DELDEL=0.0
      CK11=9.12/ASPECT+7.16*CRB
      CK12=ASPECT*CRB/2.
      CK13=(FLAPS/50.)**2.
C*********DELAL=THE CHANGE IN ALPHA AT THE PRESENT
C         ALTITUDE.
      DELAL=-CK11*CLFWB*X-(CK12/CLAWB)*OLL*CLFWB*R-CK13*DELDEL/CLAWB
      IF(DELAL.GT.0.0) DELAL=0.0
      ALGR=AL VA-DELAL
```

Figure 153 EASY XC-8A ACLS Model File For The Takeoff Simulation (Continued)

```
      A=ALGR
      GO TO 5
   40 CL1=CL
      DELCLGR=CLWBH-CL1
      ZO OL=ZO OL+DELCLGR
C  CALCULATION OF THE GROUND EFFECT UPDATE FOR
C  THE MOMENT M COEFFICIENT.
      FLAPNO=TBLU2(AL VA,CTTOT,TBCLO(7),TBCLO(4),TBCLO(17)
     1,1,1,10,3,10,3)
C*********CLWB=THE WING-BODY LIFT COEFFICIENT, FLAPS RETRACTED,
C         OUT OF GROUND EFFECT.
      CLWB=FLAPNO-SRATIO*CLH
C*********DELCLF=THE CHANGE IN LIFT COEFFICIENT DUE TO FLAPS,
C         OUT OF GROUND EFFECT.
      DELCLF=CLWBH-FLAPNO
C*********BEFF=THE EFFECTIVE WING SPAN.
      BEFF=(CLWB+DELCLF)/((CLWB/74.4)+(DELCLF/58.8))
      BEFF2=BEFF**2.
      SN1=BEFF2+4.*(HTAIL-H25CR)**2.
      SN2=BEFF2+4.*(HTAIL+H25CR)**2.
      DELE=EPSILON*SN1/SN2
      HAL=TAILA+DELE
      CLHGR=TBLU2(HAL,ELEOL,TBCLH(7),TBCLH(4),TBCLH(15)
     1,1,1,8,3,8,3)
      DELCLHG=CLHGR-CLH
C*********DELCMHG=THE CHANGE IN PITCHING MOMENT COEFFICIENT
C         PRODUCED BY THE HORIZONTAL STABILIZER. A .4MAC
C         C.G. LOCATION WAS ASSUMED.
      DELCMHG=-DELCLHG*SRATIO*4.304
      DCLWBG=DELCLGR+DELCLHG*SRATIO
C*********DCMUBG=THE CHANGE IN PITCHING MOMENT FOR THE
C         WING. A .4C C.G. LOCATION WAS ASSUMED.
      DCMWBG=-.15*DCLWBG
      DELCMG=DCMWBG+DELCMHG
      MO OL=MO OL+DELCMG
C CALCULATION OF THE GROUND EFFECTS UPDATE FOR
C  THE DRAG COEFFICIENT.
      SIGMA=EXP(-2.48*H75**.768)
      DELCDLG=SIGMA*(CLFWB**2.)/(ASPECT*3.14159)
      XO OL=XO OL+DELCDLG
  777 CONTINUE
C*******************************************************************
C*******************************************************************
LOCATION=80     FL      INPUTS=SG,VA(MAC=AMN)
LOCATION=76     FN      INPUTS=FL,TS(PT=P,2)
FORTRAN STATEMENTS
C   THE FOLLOWING STATEMENT CONVERTS THE LBS/SEC FLOW RATE TO
C   LBS/MIN. THIS VALUE IS THEN DOUBLED TO ACCOUNT FOR THE TWO
C   FANS IN THE SYSTEM.  (I.E. 60*2=120)
```

Figure 153 EASY XC-8A ACLS Model File For The Takeoff Simulation (Continued)

```
      WTRTS=120.*W2 FN
LOCATION=17     TS      INPUTS=SG,FL(PAM=PA),FN(T,2=TTR)
LOCATION=14     S2      INPUTS=
TS(FXT=FX,2,FYT=FY,2,FZT=FZ,2,TXT=TX,2,TYT=TY,2,TZT=TZ,2)
LOCATION=34     OL      INPUTS=VA,S2
LOCATION= 1     DL      INPUTS=VA,OL,S2
LOCATION=40     SG      INPUTS=OL,DL
FORTRAN STATEMENTS
      RPD=.01745329
      SP=SIN(PITSG*RPD)
      SR=SIN(ROLSG*RPD)
      CP=COS(PITSG*RPD)
      CR=COS(ROLSG*RPD)
      ACELCG=(FX2OL*SP-FY2DL*SR*CP-FZ2OL*CR*CP)/(32.174*MA1OL)-1.
      PCUSH=(PC TS-PAMFL)*144.0
      PTRUNK=(PT TS-PAMFL)*144 0
C*********ACELCG=THE VERTICAL ACCELERATION OF THE C.G.. (G)
C*********PCUSH=GAGE CUSHIOH PRESSURE (PSFG)
C*********PTRUNK=GAGE TRUNK PRESSURE (PSFG)
END OF MODEL
PRINT
```

Figure 153 EASY XC-8A ACLS Model File For The Takeoff Simulation (Concluded)

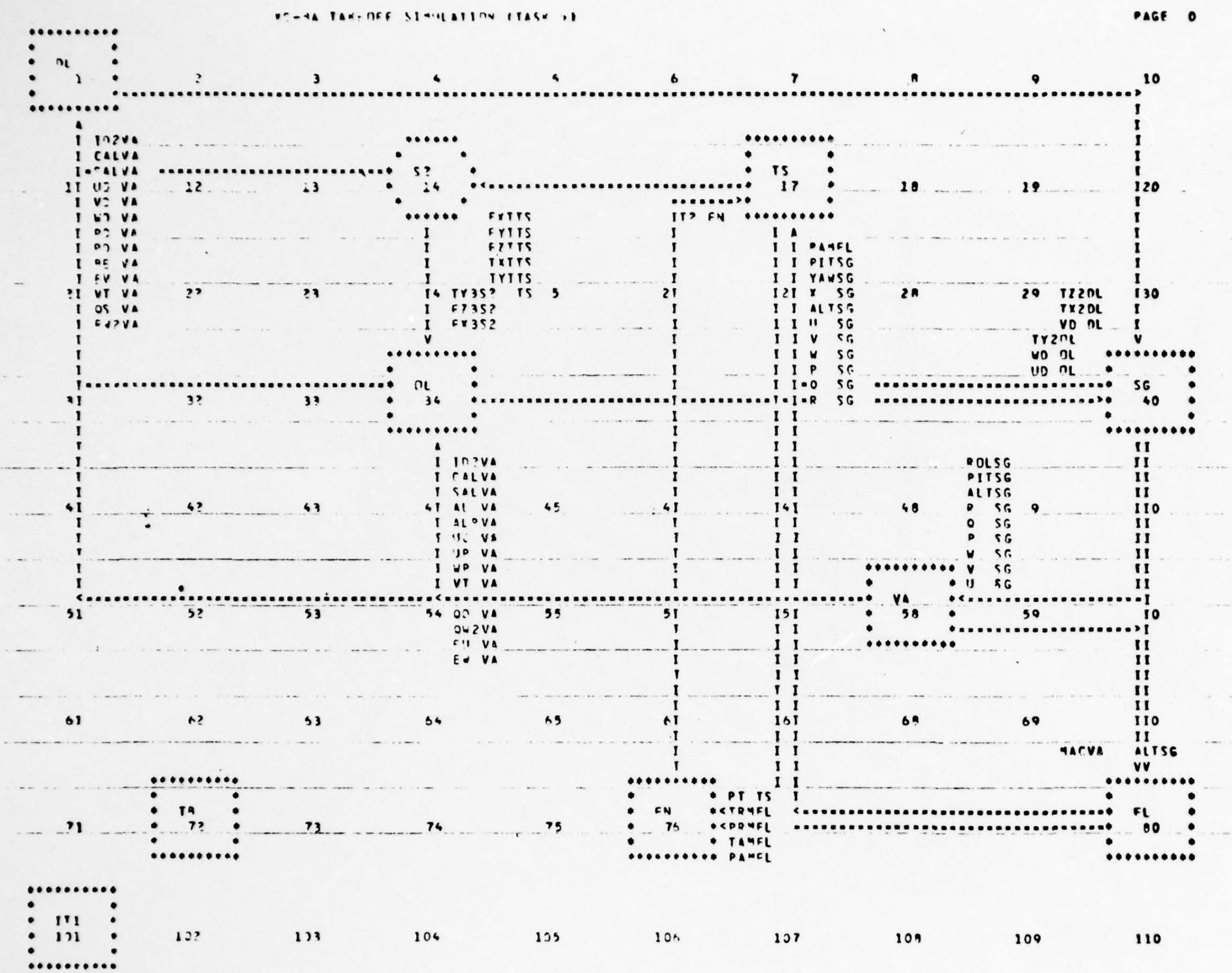


Figure 154 EASY XC-8A ACLS Model Schematic For The Takeoff Simulation

10.2.2 Analysis File

The analysis file for the XC-8A takeoff simulation is listed in Figure 155. The input data for OL, DL, SG, VA, TS, FL, FN, the ground effects, wing skid and the aerodynamic coefficient and derivative coding was taken directly from the XC-8A ACLS landing simulation analysis file (see 9.2.2). The only change required was a shift in c.g. location from 28% for the landing simulation to 28.1% for the takeoff simulation. Table B2TTB input the engine thrust schedule into the takeoff model. The thrust schedule increases linearly from 5000 to 8000 lbs/engine over the simulation time interval from 10 to 30 seconds. Since no engine thrust data was available this schedule represents an approximation of thrust schedule used during takeoff. Table B2TTB inputs the elevator schedule into the takeoff model.

A STEADY STATE command was performed to initialize the model prior to requesting the SIMULATE command. All the states except FO IT1, PC TS, VC TS, PT TS, and VT TS were frozen during the steady state analysis.

10.3 Simulation Description and Results

The EASY XC-8A model was initialized by freezing the pseudo-integral controller (IT1) along with the lateral states P SG, R SG, ROLSG, YAWSG, and V SG.

Figure 156-a, -b, -c, and -d compare the XC-8A aircraft flight test data with the EASY simulation results for aircraft pitch, cushion pressure, airspeed, and elevator deflection. The EASY simulation was initialized at 10 seconds which corresponds to 101556.0 seconds in the flight test data. As shown in Figure 152 the air cushion system becomes unstable during takeoff slide. This instability continues to increase until the aircraft takes off at approximately the 24 second mark in the computed time history. Several runs were made in an attempt to eliminate the instability without success.

```
TABLE=TBCLO,10,3
0.0,0.8,1.2
-4.0,-2.0,0.0,2.0,4.0,6.0,8.0,10.0,12.0,14.0
-.0154,.1523,.3378,.5153,.6988,.8734,1.0385
1.1809,1.3020,1.4271
-.0480,.1909,.4496,.7179,.9861,1.2423,1.4788
1.6915,1.8735,2.0650
-.0692,.2094,.4975,.7855,1.0790,1.3720,1.6264
1.8644,2.0840,2.2780
TABLE=TBCL20,10,3
0.0,0.8,1.2,
-4.0,-2.0,0.0,2.0,4.0,6.0,8.0,10.0,12.0,14.0
1.0055,1.1849,1.3435,1.4986,1.6568,1.8134
1.9626,2.0934,2.2089,2.3156
1.4464,1.7130,1.9513,2.1828,2.3979,2.6214
2.8429,3.0386,3.2056,3.3703
1.6034,1.8913,2.1564,2.3988,2.6470,2.8904
3.1287,3.3488,3.5450,3.7086
TABLE=TBCL40,10,3
0.0,0.8,1.2
-4.0,-2.0,0.0,2.0,4.0,6.0,8.0,10.0,12.0,14.0
1.6639,1.8323,1.9766,2.1411,2.3094,2.4749
2.5999,2.7305,2.8406,2.9339
2.4030,2.6580,2.8879,3.1195,3.3492,3.5917
3.7955,3.9699,4.1346,4.2591
2.6590,2.9325,3.1828,3.4399,3.7065,3.9480,4.1946
4.4070,4.5844,4.7244
TABLE=TBX,11
0.0,.1,.2,.3,.4,.5,.6,.7,.8,.9,1.0
1.0334,.7145,.5360,.4176,.3239,.2586,.2080
.1669,.1352,.1087,.0899
TABLE=TBCLH,8,3
-25.0,0.0,15.0
-18.0,-16.0,-14.0,8.0,10.0,12.0,14.0,16.0
-1.6755,-1.6289,-1.5699,-.3587,-.2409
-.1127,.00755,.1004
-1.0937,-1.0272,-.9288,.4601,.5818
.7110,.7566,.6414
-.6462,-.5369,-.4162,1.0091,1.1333,1.0773
.9537,.8311
TABLE=TBLLO,12,3
10.0,18.0,24.0
.2,.4,.6,.8,1.0,1.2,1.4,1.6,1.8,2.0,2.2,2.4
.2118,.06113,.00509,-.01720,-.02592,-.02780,-.02880,-.02938
-.02902,-.02856,-.02814,-.02824
-.05664,-.09657,-.1098,-.1063,-.09462,-.08550,-.07764,-.07006
-.06439,-.06119,-.05832,-.05740
-.1942,-.1980,-.1855,-.1648,-.1426,-.1250,-.1114
-.09891,-.08937,-.08117,-.07524,-.07178
```

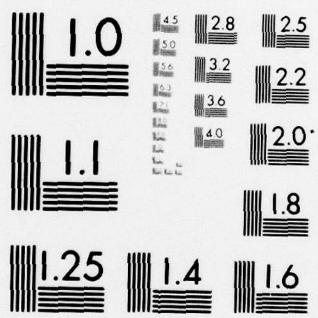
Figure 155 EASY XC-8A ACLS Analysis File For The Takeoff Simulation

AD-A080 489 BOEING AEROSPACE CO SEATTLE WA BOEING MILITARY AIRPL--ETC F/G 9/2
EASY ACLS DYNAMIC ANALYSIS. VOLUME III. DESCRIPTION OF SIMULATI--ETC(U)
SEP 79 M K WAHI, P R PERKINS, G S DULEBA F33615-77-C-3054

UNCLASSIFIED AFFDL-TR-79-3105-VOL-3 NL

4 OF 4
AD
A080489

END
DATE
FILMED
3 - 80
DDC



MICROCOPY RESOLUTION TEST CHART

NATIONAL BUREAU OF STANDARDS-1963-A

```
TABLE=TBDEL,5
.2,.4,.6,.8,1.0
-.09898,-.06246,-.03709,-.01953,-.005234
TABLE=ABLTS,9
71.9,0.0,71.9,.05,0.0,.5
47.1,10.0,57.1,.12,0.0,.5
44.2,15.0,56.8,.17,0.0,.3
TABLE=XYZTS,32
148.79,1.6,62.5,80.0
147.69,4.5,62.5,60.0
145.69,6.9,62.5,40.0
142.29,8.7,62.5,15.0
124.44,9.0,62.5,0.0
93.44,9.0,62.5,0.0
62.44,9.0,62.5,0.0
29.29,9.0,62.5,0.0
5.79,9.0,62.5,0.0
-17.96,9.0,62.5,0.0
-41.46,9.0,62.5,0.0
-65.21,9.0,62.5,0.0
-83.91,8.8,62.5,-12.5
-87.21,7.4,62.5,-35.0
-89.61,4.8,62.5,-57.5
-90.91,1.6,62.5,-80.0
TABLE=DM TS,16
20.0,.097,20.0,.097,20.0,.097,30.0,.097
31.0,.097,31.0,.097,31.0,.097,34.0,.097
13.25,.097,34.0,.097,13.25,.097,34.0,.097
22.5,.097,22.5,.097,25.0,.097,20.0,.097
TABLE=IALTS,32
1,.02189,21.0,23.0
1,.02189,21.0,23.0
2,.01911,22.5,14.0
2,.01911,22.5,14.0
3,.01911,22.25,14.0
3,.01911,22.25,14.0
3,.01911,22.25,14.0
3,.0025,19.5,21.0
3,.01911,23.5,13.0
3,.0025,19.5,21.0
3,.01911,23.5,13.0
3,.0025,19.5,21.0
2,.01911,22.5,14.0
2,.01911,22.5,14.0
1,.02189,21.0,23.0
1,.02189,21.0,23.0
TABLE=ENDTS,2
9.0,.45,9.0,.45
TABLE=SPHTS,8,3
```

Figure 155 EASY XC-8A ACLS Analysis File For The Takeoff Simulation (Continued)

```
1,2,3
0.0,33.,66.,100.,150.,200.,300.,500.
0.0,.2288,.6903,1.0079,1.1108,1.1963,1.3127,1.4880
0.0,.7809,1.8073,2.1367,2.2924,2.4125,2.5653,2.76
0.0,.9804,2.2110,2.5446,2.7195,2.8521,3.0180,3.2197
TABLE=STHTS,8,3
1,2,3
0.0,33.,66.,100.,150.,200.,300.,500.
0.0,.9804,2.2110,2.5446,2.7195,2.8521,3.0180,3.2197
0.0,.9804,2.2110,2.5446,2.7195,2.8521,3.0180,3.2197
0.0,.0135,.02702,.04094,.0614,.08187,.1228,.16374
TABLE=PM TS,14
7,0.0,0.0,.9
8,.8,5.0,.6
9,0.0,0.0,.8
10,.8,5.0,.6
11,0.0,0.0,.8
12,.8,5.0,.6
13,0.0,0.0,.9
TABLE=RELTS,2
0.0,50.0
0.0,0.0
TABLE=BWTTS,4
344.1,160.0,344.1,160.0,0,0,0,0
TABLE=FANFN,16,3
5260.,5880.,6370.
1.0,1.051,1.075,1.100,1.126,1.150,1.175,1.195
1.201,1.221,1.242,1.251,1.279,1.289,1.300,1.339
62.56,60.64,59.11,57.0,54.14,50.66,45.82,39.37
-.203,-1.444,-2.615,-3.158,-4.715,-5.258,-5.897,-8.153
69.18,67.8,66.81,65.22,63.39,61.3,58.74,56.25,55.48
51.73,45.52,-.275,-1.939,-2.483,-3.05,-5.353
74.87,73.67,72.99,72.09,70.92,69.55,67.97,66.25
65.77,63.56,60.81,59.16,50.28,-.231,-.763,-2.923
TABLE=STAFN,17
1.0,1.021,1.041,1.061,1.081,1.100,1.120,1.140
1.16,1.181,1.201,1.222,1.241,1.260,1.281,1.30,1.312
0.0,7.0,12.54,17.19,21.23,25.01,28.45,31.54
34.57,37.42,40.05,42.95,45.31,47.82,50.36,52.82,54.17
TABLE,A2TTB,2
10.0,30.0
8000.0,8000.0
TABLE,B2TTB,5
10.0,16.0,26.0,28.0,30.0
-25.0,-17.91,-12.0,-15.76,-10.7
PARAMETER VALUES
NUIFN=.2,NUFFN=.87
CORFN=1.0,RPMFN=6186.
FSENG=197.0,BLENG=183.0,WLENG=191.0
```

Figure 155 EASY XC-8A ACLS Analysis File For The Takeoff Simulation (Continued)

```
FSCG=343.53,BLCG=0.0,WLCG=160.0
FS75CR=406.81,WL75CR=206.06,FS25CR=336.79
WL25CR=206.06,FSTAIL=889.65,WLTAIL=403.84
ROOTC=11.67,ASPECT=9.75,SH=233.0
THETA=0.0,TRUNK=1.0
CGRAV=28.1,SPOIL=0.0,FLAPS=30.0
ID1VA=3,VS VA=27.37,ALSVA=1.0
S  VA=945.0,IDGVA=6
WCUTS=0.0,TCUTS=560.0
ANETS=-16,CDGTS=1.0,CD1TS=.62
CD2TS=.2,TAUTS=.005,DMPTS=.02
EPCTS=1.
VU TS=6.0,PTMTS=3.0
SPBTS=0.0
MA1OL=1114.,C  OL=10.29,ISWOL=3
B  DL=96.0
RUDDL=0.0,AILDL=0.0
IXXSG=183700.,IYYSG=228200.
IZZSG=365100.,IXZSG=29220.
FSSKID=340.0,BLSKID=432.5,WLSKID=90.0
SPLOW=2666.,SPHIGH=50000.,DEFMX=1.875
SKIDMU=.02,SKID=1.0
GKIIT1=10.0
INITIAL CONDITIONS
U  SG=27.36583,V  SG=0.0,W  SG=.47767
P  SG=0.0,Q  SG=0.0,R  SG=0.0
ROLSG=0.0,PITSG=1.0,YAWSG=0.0
ALTSG=8.0,X  SG=0.0,Y  SG=0.0
PT TS=16.627,VT TS=850.29
PC TS=14.731,VC TS=139.47
FO IT1=150.0
ERROR CONTROLS
U  SG=.001,V  SG=.0001,W  SG=.001
P  SG=.0001,Q  SG=.0001,R  SG=.0001
ROLSG=.0001,PITSG=.0001,YAWSG=.0001
ALTSG=.0001,X  SG=.0001,Y  SG=.0001
PT TS=.0001,VT TS=.001
PC TS=.0001,VC TS=.001
FO IT1=.0001
INITIAL TIME=10.0
PRINT CONTROL=4
SS ITERATIONS=60
NO STATES
INT CONTROLS,PC TS=1,VC TS=1
PT TS=1,VT TS=1,FO IT1=1
LINEAR ANALYSIS
STEADY STATE
XIC-X
LINEAR ANALYSIS
```

Figure 155 EASY XC-8A ACLS Analysis File For The Takeoff Simulation (Continued)

```
TINC=.1,TMAX=30.0,INT MODE=6
ALL STATES
INT CONTROLS,FO IT1=0
P  SG=0,R  SG=0
ROLSG=0,YAWSG=0,V  SG=0
LINEAR ANALYSIS
PRINTER PLOTS,PLOT ON
DISPLAY1
PCUSH,VS,TIME
PTRUNK,VS,TIME
VC TS,VS,TIME
VT TS,VS,TIME
DISPLAY2
ROLSG,VS,TIME
PITSG,VS,TIME
YAWSG,VS,TIME
Q  SG,VS,TIME
DISPLAY3
X  SG,VS,TIME
Y  SG,VS,TIME
ALTSG,VS,TIME
ACELCG,VS,TIME
VKNOTS,VS,TIME
DISPLAY4
ELEOL,VS,TIME
TITLE      XC-8A TAKEOFF SIMULATION
PLOT ID      PAUL R. PERKINS  MS 47-03
SIMULATE
XIC-X
LINEAR ANALYSIS
```

Figure 155 EASY XC-8A ACLS Analysis File For The Takeoff Simulation (Concluded)

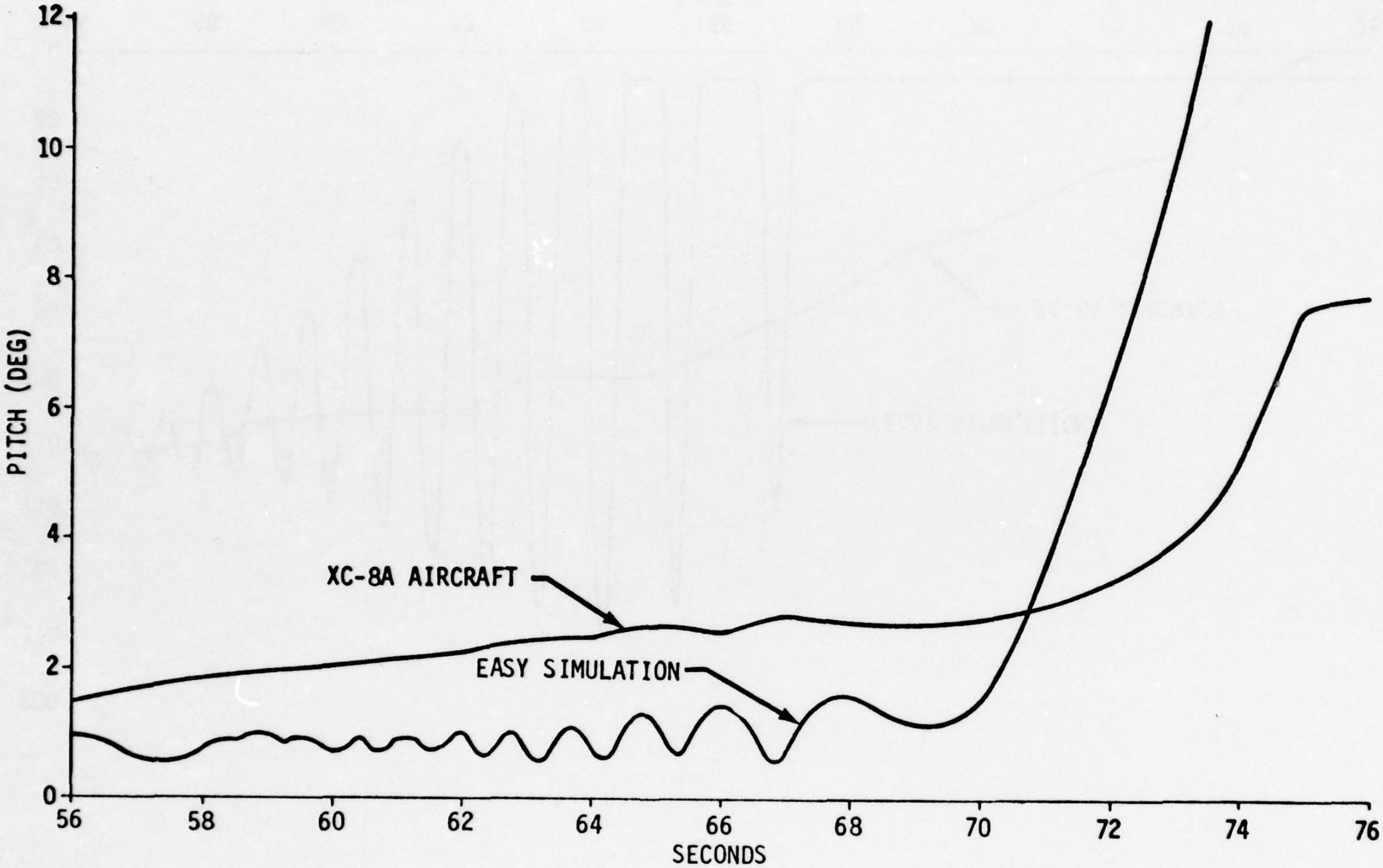


Figure 156-(a) XC-8A Takeoff Simulation A Comparision of the XC-8A Aircraft with the EASY Simulation Results

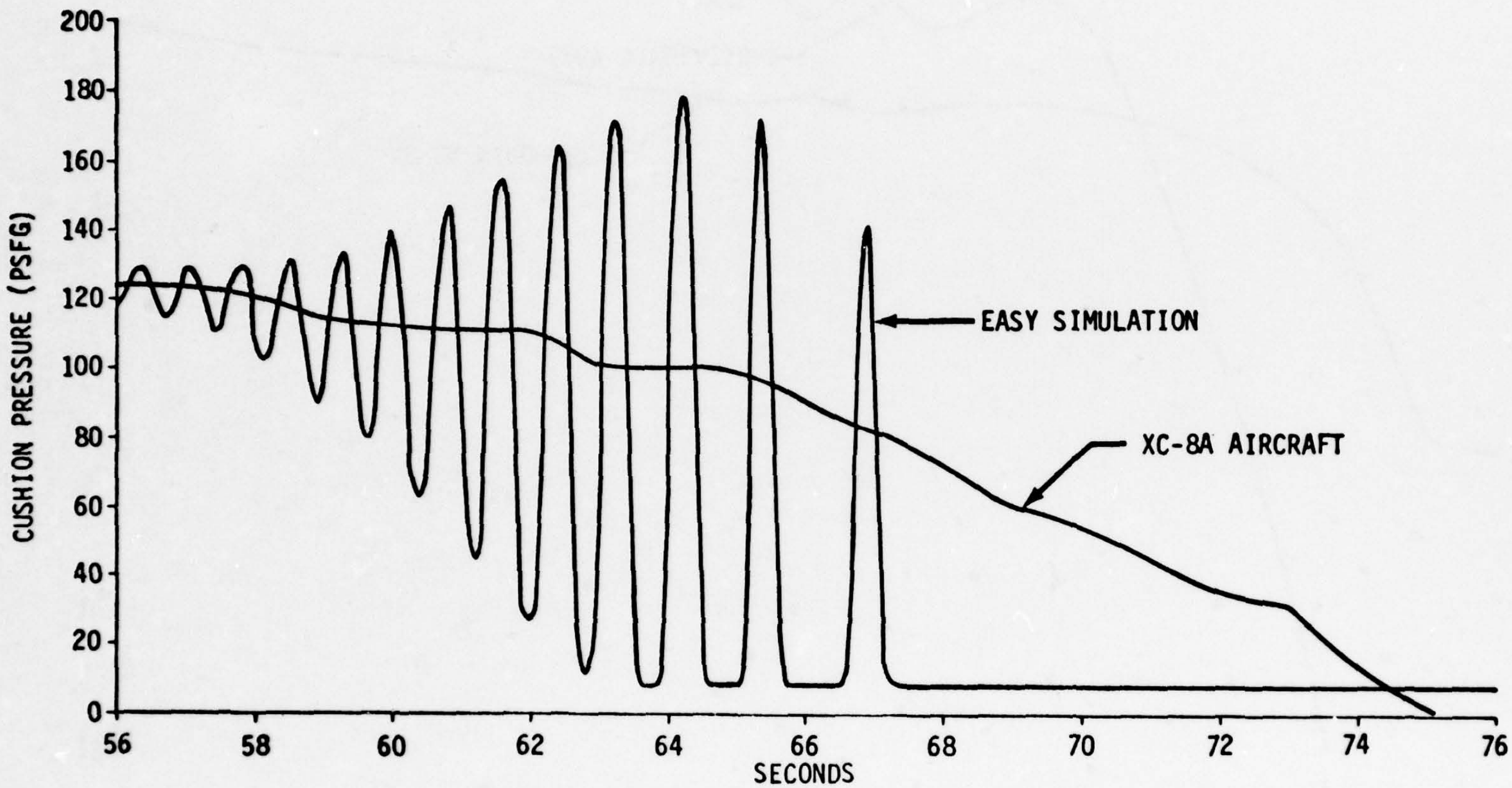


Figure 156-(b) XC-8A Takeoff Simulation A Comparision of the XC-8A Aircraft with the EASY Simulation Results

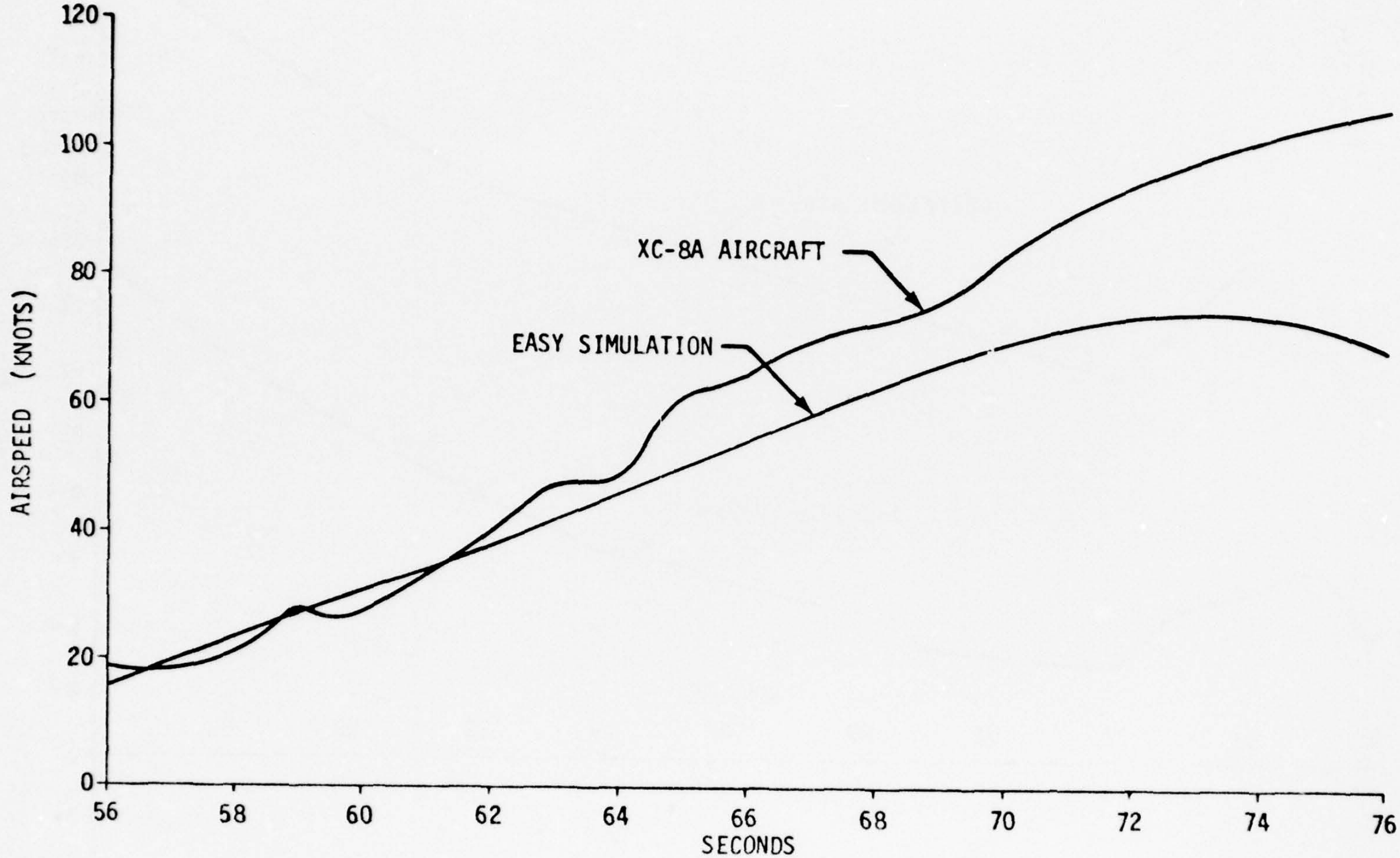


Figure 156-(c) XC-8A Takeoff Simulation A Comparision of the XC-8A Aircraft with the EASY Simulation Results

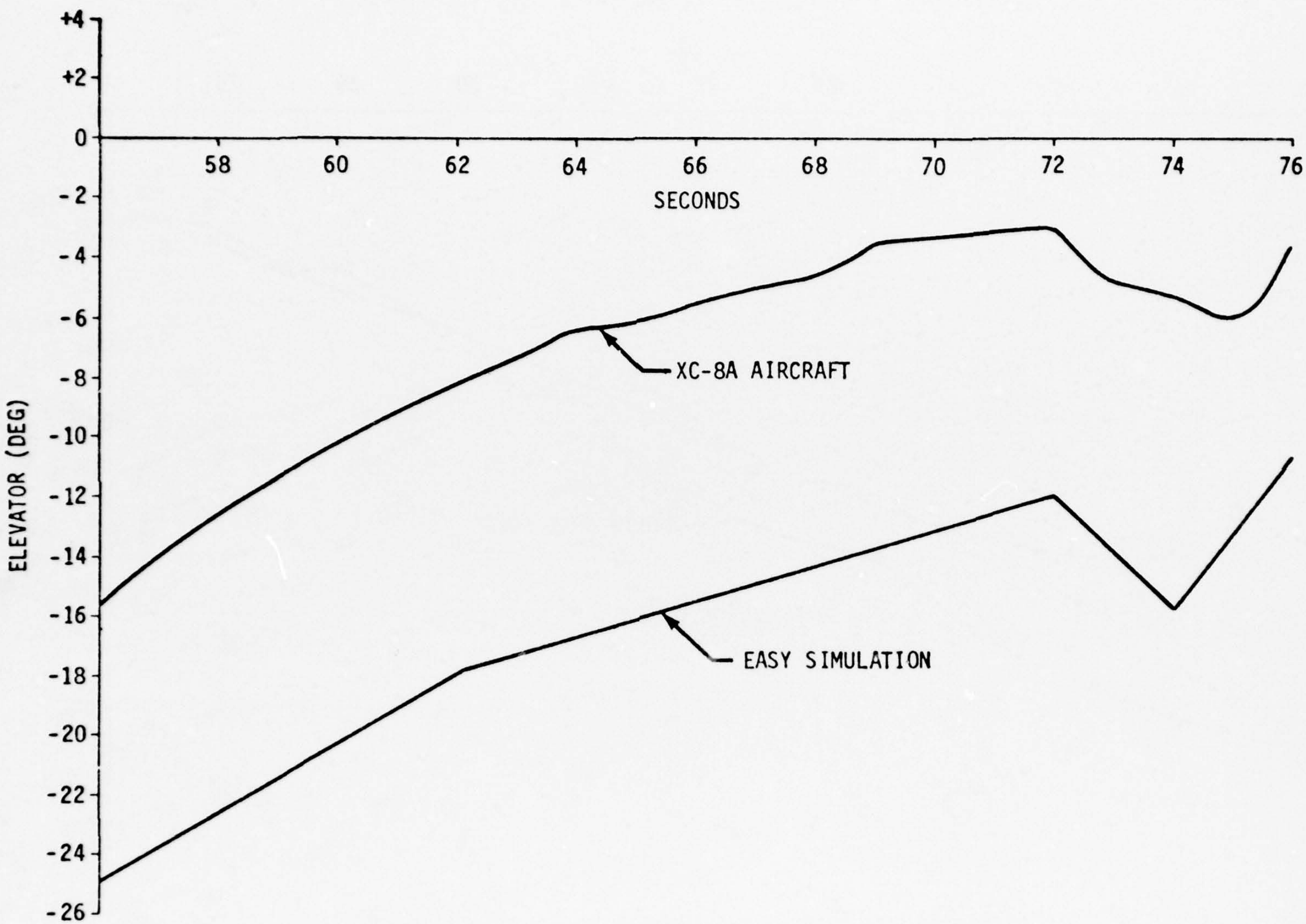


Figure 156-(d) XC-8A Takeoff Simulation A Comparision of the XC-8A Aircraft with the EASY Simulation Results

10.4 Discussion

There are three possible causes for the ACLS/vehicle instability experienced during the simulation of the XC-8A takeoff slide. They are:

1. A numerical instability associated with the integration algorithm. Typically a numerical instability would generate random sharp fluctuations in the computer time histories. Since the EASY simulation time histories demonstrated a uniform periodic oscillation the cause of the instability is probably not numerical in origin.

2. A hardware instability. It is a real possibility that the trunk-cushion system is unstable under certain operating conditions. Unstable pitch and plunge mode oscillations did occur during the flight testing of the XC-8A ACLS. Subtle differences between the hardware and the simulation of the hardware, such as the nature of trunk and ground friction, may result in marginally stable hardware being simulated as an unstable system.

3. A model instability associated with the mathematical representation of hardware components.

10.5 Conclusions

Verification of the EASY XC-8A simulation was not accomplished due to the inability to resolve the system slideout instability within the program schedule.

SECTION XI

XC-8A TAXI SIMULATION

11.1 Objectives

The objective was to simulate a typical XC-8A taxi excursion by modifying the EASY XC-8A ACLS model created for the landing simulation. The EASY XC-8A taxi simulation was compared with actual flight test data to validate the computer model.

11.2 Technical Approach

A taxi excursion down a 2 foot high, 23 ft long ramp from flight test #F033175-1 was selected for simulation using the EASY XC-8A ACLS model. A complete listing of the taxi configuration is listed in Table 14.

11.2.1 Model File

Thirteen standard EASY components and several pages of Fortran coding were assembled to form the EASY XC-8A ACLS taxi simulation model. The EASY XC-8A model file and computer generated model schematic for the taxi simulation are given in Figures 157 and 158. The aerodynamics and the ACLS components remain unchanged from the landing simulation model file (See 9.2.1). There are five pseudo-integral controllers (See Appendix B) implemented to initialize the taxi model. These controllers are:

- IT1 varies the engine thrust until the total airspeed is 11.822 ft/sec
- IT2 varies the elevator until q is zero
- IT3 varies the rudder deflection until the v velocity is zero
- IT4 varies the aileron deflection until the p angular rate is zero
- IT5 varies the trunk to cushion trim valve area (CAVTS) until the trunk pressure is 16.9072 psia.

TABLE 14 XC-8A TAXI CONFIGURATION

(TIME=20 SEC)

FLAPS	7.0 DEG
TRUE AIRSPEED	12.16 FT/SEC
ALTITUDE (c.g.)	9.34 FT
GROSS WEIGHT	35000 LBS
CENTER OF GRAVITY (c.g.)	28.0% MAC
FSCG	343.41 IN
WLCG	160.0 IN
MOMENTS OF INERTIA:	
IXX	183700 SLUGS-FT^2
IYY	228200 SLUGS-FT^2
IZZ	365100 SLUGS-FT^2
IXZ	29220 SLUGS-FT^2
THRUST:	
THRRT	1425.1 LBS
THRLT	1425.1 LBS
ATTITUDE:	
ANGLE OF ATTACK (α)	3.19 DEG
SIDESLIP ANGLE (β)	0.0 DEG
FLIGHT PATH ANGLE (γ)	0.0 DEG
ROLL (ϕ)	0.0 DEG
PITCH (θ)	3.19 DEG
YAW (ψ)	0.0 DEG
LONGITUDINAL CONTROL SURFACE DEFLECTIONS:	
ELEVATOR	-25.0 DEG
SPOILER	0.0 DEG
LATERAL CONTROL SURFACE DEFLECTIONS:	
RUDDER	-.01 DEG
AILERON	3.54 DEG

TABLE 14 XC-8A TAXI CONFIGURATION (CONCLUDED)

(TIME=20 SEC)

AIR CUSHION PARAMETERS	
TRUNK PRESSURE........................	320.0 PSFG
CUSHION PRESSURE......................	133.5 PSFG
FAN RPM...............................	5260 RPM
EFFECTIVE AREA FOR TRUNK TO CUSHION	
TRIM VALVE............................	144.77 IN^2

```
 MODEL DESCRIPTION    XC-8A TAXI SIMULATION (TASK 5)
FORTRAN STATEMENTS
      COMMON /BMADTS/IERR
      REAL L1,M1,N1
      DATA IERR/0/
ADD PARAMETERS=THETA
FSENG,BLENG,WLENG,FSCG,BLCG,WLCG
FSSKID,BLSKID,WLSKID,SPLOW,SPHIGH,DEFMX,SKIDMU,SKID
TRUNK,CGRAV,FLAPS,SPOIL
ROOTC,ASPECT,SH,FS25CR,WL25CR
FS75CR,WL75CR,FSTAIL,WLTAIL
ADD VARIABLES=CTRT,CTLT,CTTOT,FFBETA
THRRT,THRLT
PCUSH,PTRUNK
COMP,ZFORCE,XFORCE
DELCLGR,DELCDLG,DELCMG
ACELCG,VKNOTS
ADD TABLES
TBCL0,46,TBCL20,46,TBCL40,46
TBCLH,38,TBX,-25,TBLL0,54,TBDEL,-13
LOCATION=58    VA    INPUTS=SG
FORTRAN STATEMENTS
      FINIT1=VT VA-11.822
      FINIT2=Q  SG
      FINIT3=V  SG
      FINIT4=P  SG
      FINIT5=16.9072-PT TS
LOCATION=101    IT1
LOCATION=103    IT2
LOCATION=105    IT3
LOCATION=107    IT4
LOCATION=109    IT5
FORTRAN STATEMENTS
THRRT=F0 IT1
      THRLT=THRRT
      ELEOL=F0 IT2
      RUDDL=F0 IT3
      AILDL=F0 IT4
      CAVTS=F0 IT5
C*********VKNOTS=THE TRUE AIR SPEED. (KNOTS)
      VKNOTS=VT VA/1.6889
C***********************************************************************
C***********************************************************************
C  THESE FORTRAN STATEMENTS UPDATE THE COS(ALPHA) AND SIN(ALPHA)
C  WITHOUT CHANGING THE VALUE OF ALSVA. THIS PROVIDES FOR CORRECT
C  RESOLUTION OF THE LIFT AND DRAG FORCES FROM THE STABILITY
C  AXES INTO THE BODY AXES. THESE FIVE FORTRAN STATEMENTS
C  MUST BE IMPLEMENTED WHENEVER THE AERODYNAMIC
C  COEFFICIENTS OR DERIVATIVES ARE INPUT AS FUNCTIONS
```

Figure 157 EASY XC-8A ACLS Model File For The Taxi Simulation

```
C  OF THE ANGLE OF ATTACK.
C  CAUTION- THESE FORTRAN STATEMENTS CANNOT BE USED WHEN THE
C  *******  AERODYNAMIC DERIVATIVES ARE REFERENCED TO THE BODY AXIS.
      RPD=.01745329
      SALVA=SIN(AL VA*RPD)
      CALVA=COS(AL VA*RPD)
      PO VA=P  SG*CALVA+R  SG*SALVA
      RO VA=R  SG*CALVA-P  SG*SALVA
C*********************************************************************
C*********************************************************************
C THE FOLLOWING IF STATEMENTS LIMIT THE ALLOWABLE DEFLECTIONS
C OF THE ELEVATOR,RUDDER,FLAPS AND AILERION WHEEL. THEY ALSO
C LIMIT THE ENGINE THRUST TO A SPECIFIED RANGE.
C*********RANGE OF ELEVATOR DEFLECTION= -25.0 TO +15.0 DEG.
C          (A POSITIVE ELEVATOR DEFLECTION IS DOWN)
      IF(ELEOL.LT.-25.0) ELEOL=-25.0
      IF(ELEOL.GT.15.0) ELEOL=15.0
C*********RANGE OF RUDDER DEFLECTION= -20.0 TO +20.0 DEG.
C          (A POSITIVE RUDDER DEFLECTION IS TO THE PILOTS LEFT)
      IF(RUDDL.LT.-20.0) RUDDL=-20.0
      IF(RUDDL.GT.20.0) RUDDL=20.0
C*********RANGE OF FLAP DEFLECTION= 0.0 TO +40.0 DEG.
      IF(FLAPS.LT.0.0) FLAPS=0.0
      IF(FLAPS.GT.40.0) FLAPS=40.0
C*********RANGE OF AILERION WHEEL ROTATION= -75.0 TO +75.0 DEG.
C          (A POSITIVE AILERION DEFLECTION TIPS THE RIGHT WING DOWN)
      IF(AILDL.LT.-75.0) AILDL=-75.0
      IF(AILDL.GT.75.0) AILDL=75.0
C*********THE MAXIMIUM THRUST PER ENGINE=+9000.0 LBS..
      IF(THRRT.GT.9000.0) THRRT=9000.0
      IF(THRLT.GT.9000.0) THRLT=9000.0
C*********THE MAXIMIUM REVERSE THRUST PER ENGINE=-9000.0
      IF(THRRT.LT.-9000.0) THRRT=-9000.0
      IF(THRLT.LT.-9000.0) THRLT=-9000.0
C*********************************************************************
C*********************************************************************
C  THESE FORTRAN STATEMENTS CALCULATE THE BODY AXIS FORCES
C  AND MOMENTS PRODUCED BY THE ENGINE THRUST. THE THRUST
C  LOCATION AND ANGLE OF THRUST VECTOR MUST BE SPECIFIED.
C  THE MODEL CAN ACCEPT REVERSE THRUST AND/OR DIFFERENTIAL
C  THRUST SCHEDULES.
C*********XENG,YENG AND ZENG=THE XYZ BODY AXIS MOMENT ARM
C          VECTORS (FEET)
C  THE REQUIRED INPUT PARAMETERS ARE
C*********THRLT=LEFT ENGINE THRUST(LBS.)
C*********THRRT=RIGHT ENGINE THRUST(LBS.)
C*********THETA=THE ANGLE OF THRUST APPLICATION IN THE XZ PLANE
C           (DEGREES)
C*********FSENG,BLENG,WLENG. THESE ARE THE THE FUSELAGE
```

Figure 157 EASY XC-8A ACLS Model File For The Taxi Simulation (Continued)

```
C         STATION,BUTT LINE, AND WATER LINE LOCATIONS
C         OF THE RIGHT ENGINE THRUST VECTOR. (IN.)
C*********FSCG,BLCG,WLCG. THESE ARE THE FUSELAGE STATION,
C         BUTT LINE AND WATER LINE LOCATIONS OF THE C.G..(IN.)
      RPD=.01745329
      FLPRAD=FLAPS*RPD
      THRAD=THETA*RPD
      COSINE=COS(THRAD)
      SINE=SIN(THRAD)
      XENG=(FSCG-FSENG)/12.0
      YENG=(BLENG-BLCG)/12.0
      ZENG=(WLCG-WLENG)/12.0
      THDIF=THRLT-THRRT
      THRT=THRRT
      THLT=THRLT
      IF(THRT.LT.0.0) THRT=0.0
      IF(THLT.LT.0.0) THLT=0.0
      DIF=THLT-THRT
      FX1=(THRLT+THRRT)*COSINE
      FY1=(.002+.0183*FLPRAD)*DIF
      FZ1=-(THRLT+THRRT)*SINE
      T1=-.0176*(THRLT+THRRT)
      T2=(.0125+.1095*FLPRAD)*DIF
      TX1=T1+T2
      TY1=FX1*ZENG-FZ1*XENG
      TZ1=(.1575-.0322*FLPRAD)*THDIF
C*********************************************************************
C*********************************************************************
C THESE FORTRAN STATEMENTS DETERMINE THE BODY AXIS FORCES AND
C TORQUES PRODUCED WHEN A WING SKID TOUCHES THE GROUND.THE REQUIRED
C INPUT PARAMETERS ARE
C*********FSSKID,BLSKID,WLSKID ARE THE FUSELAGE STATION,BUTT LINE
C         AND WATER LINE LOCATIONS OF THE TIP OF THE UNDEFORMED
C         RIGHT SKID. IT IS ASSUMED THAT THE SKIDS ARE AT SYMMETRIC
C         LOCATIONS W.R.T. THE X-Z PLANE. (IN.)
C*********FSCG,BLCG,WLCG. THESE ARE THE FUSELAGE STATION,
C         BUTT LINE AND WATER LINE LOCATIONS OF THE C.G.. (IN.)
C*********SPLOW=LOW SPRING CONSTANT OF THE SKID (I.E. NORMAL SPRING
C         RATE).(LBS/FT)
C*********SPHIGH=HIGH SPRING CONSTANT WHICH REPRESENTS THE SPRING
C         CONSTANT OF THE BOTTOMED OUT SKID.(LBS/FT)
C*********DEFMX=THE MAXIMIUM NORMAL DEFLECTION OF THE SKID.(FT)
C         DEFLECTIONS BEYOND DEFMX ARE CONTROLLED BY THE
C         HIGHER SPRING RATE.
C*********SKIDMU=THE COEFFICIENT OF DYNAMIC FRICTION BETWEEN
C         THE SKID AND THE GROUND.
C*********SKID-THE SKID IS ELIMINATED FROM THE CALCULATIONS
C         WHEN SKID=0.0.
      FX2=0.0
```

Figure 157 EASY XC-8A ACLS Model File For The Taxi Simulation (Continued)

```
      FY2=0.0
      FZ2=0.0
      TX2=0.0
      TY2=0.0
      TZ2=0.0
      COMP=0.0
      XFORCE=0.0
      ZFORCE=0.0
      IF(SKID.EQ.0.0) GO TO 11
      XSKID=(FSCG-FSSKID)/12.0
      YSKID=(BLSKID-BLCG)/12.0
      ZSKID=(WLCG-WLSKID)/12.0
      IF(ROLSG.LT.0.0) YSKID=-YSKID
      RPD=.01745329
      CP=COS(PITSG*RPD)
      SP=SIN(PITSG*RPD)
      CR=COS(ROLSG*RPD)
      SR=SIN(ROLSG*RPD)
      CY=COS(YAWSG*RPD)
      SY=SIN(YAWSG*RPD)
C*********HGT IS THE HEIGHT OF THE SKID ABOVE THE GROUND. IF HGT
C         IS NEGATIVE THE SKID SPRING IS COMPRESSED.
      HGT=+XSKID*SP-YSKID*SR*CP-ZSKID*CR*CP+ALTSG
      IF(HGT.GE.0.0) GO TO 11
      COMP=-HGT/(SQRT(1.0-(SR*CP)**2.))
C*********COMP IS THE AMOUNT THAT THE SKID SPRING IS COMPRESSED.
C         COMP DOES NOT EQUAL HGT BECAUSE THE AIRCRAFT IS ROTATED
C         W.R.T. THE EARTH. THE DENOMINATOR OF THIS ASSIGNMENT
C         STATEMENT IS THE COSINE OF THE ANGLE WHICH THE SKID
C         MAKES FROM VERTICAL.
      HIGH=SPHIGH
      IF(COMP.LT.DEFMX) HIGH=SPLOW
C         THE EARTH AXIS FORCES PRODUCED BY THE COMPRESSION OF
C         THE SKID SPRING ARE DETERMINED BELOW.
      ZFORCE=-(DEFMX*SPLOW+(COMP-DEFMX)*HIGH)
      XFORCE=SKIDMU*ZFORCE
      IF(IERR.NE.1) GO TO 211
      WRITE(6,210)
  210 FORMAT(10X,35HA WING SKID IS TOUCHING THE GROUND.)
  211 CONTINUE
C         THE BODY AXIS FORCES ARE CALCULATED BELOW.
      FX2=XFORCE*CP*CY-ZFORCE*SP
      FY2=XFORCE*(SR*SP*CY-CR*SY)+ZFORCE*SR*CP
      FZ2=XFORCE*(CR*SP*CY+SR*SY)+ZFORCE*CR*CP
C         THE BODY AXIS TORQUES ARE CALCULATED BELOW.
      TX2=YSKID*FZ2-ZSKID*FY2
      TY2=ZSKID*FX2-XSKID*FZ2
      TZ2=XSKID*FY2-YSKID*FX2
   11 CONTINUE
```

Figure 157 EASY XC-8A ACLS Model File For The Taxi Simulation (Continued)

```
C THE NEXT 6 STATEMENTS SUM THE FORCES AND MOMENTS PRODUCED
C BY THE ENGINES AND SKIDS.
      FX1S2=FX1+FX2
      FY1S2=FY1+FY2
      FZ1S2=FZ1+FZ2
      TX1S2=TX1+TX2
      TY1S2=TY1+TY2
      TZ1S2=TZ1+TZ2
C*********************************************************************
C*********************************************************************
C THESE FORTRAN STATEMENTS DETERMINE THE AERODYNAMIC
C DERIVATIVES AND COEFFICIENTS USED IN COMPONENTS OL AND DL.
C THE CALCULATED DERIVATIVES AND COEFFICIENTS ARE ALL WITH
C RESPECT TO THE STABILITY AXIS AND ARE ALL NONDIMENSIONAL
C VALUES. THE MOMENT TERMS ARF FOR A C.G. LOCATION AT 40
C PERCENT MAC. XP1OL CORRECTS THE DATA FOR A DIFFERENT
C C.G. LOCATION.
C THE XC-8A AERODYNAMIC MODEL WAS OBTAINED FROM G.KURYLOWICH
C AIR FORCE FLIGHT DYNAMICS LABORATORY.
C         ********************************************
C         * THE FOLLOWING VARIABLES AND PARAMETERS   *
C         * ARE NEEDED TO DETERMINE THE AERODYNAMIC  *
C         * COEFFICIENTS AND DERIVATIVES. THESE ARE  *
C         * NOT EASY STANDARD COMPONENT VARIABLES OR *
C         * PARAMETERS.                              *
C         *   TRUNK-  THE TRUNK IS DEFLATED WHEN     *
C         *         TRUNK=0.0, AND FULLY INFLATED    *
C         *         WHEN TRUNK=1.0.                  *
C         *   CGRAV-  THE LOCATION OF THE C.G.. IT   *
C         *         IS  INPUT AS A PERCENT MAC.      *
C         *   FLAPS-  THE ANGULAR DEFLECTION OF THE  *
C         *         FLAPS,INPUT IN DEGREES.          *
C         *   SPOIL-  THE FRACTION OF THE TOTAL      *
C         *         SPOILER DEFLECTION. THE RANGE    *
C         *         OF SPOIL IS FROM 0.0 TO 1.0. THE *
C         *         SPOILERS ARE FULLY DEPLOYED WHEN *
C         *         SPOIL=1.0, AND FULLY RETRACTED   *
C         *         WHEN SPOIL=0.0.                  *
C         ********************************************
C THIS NEXT GROUPING OF FORTRAN STATEMENTS EVALUATES
C FREQUENTLY USED EXPRESSIONS PRESENT IN THE AERODYNAMIC
C EQUATIONS.
C*********CTRT IS THE RIGHT ENGINE COEFFICIENT OF THRUST
C*********CTLT IS THE LEFT ENGINE COEFFICIENT OF THRUST.
C*********CTTOT IS THE TOTAL COEFFICIENT OF THRUST.
      CTRT=THRRT/QS VA
      CTLT=THRLT/QS VA
      CTTOT=CTRT+CTLT
C*********IF CTTOT IS NEGATIVE,CTTOT IS THEN SET TO ZERO.
```

Figure 157 EASY XC-8A ACLS Model File For The Taxi Simulation (Continued)

```
      IF(CTTOT.LT.0.0) CTTOT=0.0
      CTT2=CTTOT**2.
      RPD=.01745329
      ALRAD=AL VA*RPD
      BERAD=BE VA*RPD
      COS2BE=(COS(BERAD))**2.
      SIN2BE=(SIN(BERAD))**2.
      BEABS=ABS(BERAD)
      FFBETA=ABS(1.-.955*(BEABS-.523))
      IF(BEABS.LE..523) FFBETA=1.0
      FLPRAD=FLAPS*RPD
      FLP2=FLPRAD**2.
      ABZF=(WLCG-160.)/12.0
      ABZC=(WLCG-74.6)/12.0
      PIH=3.141592654/2.0
C AERODYNAMIC CONTRIBUTIONS FOR THE APUS,DIFFUSERS AND CUSHION.
C ( A 40 PERCENT C.G. LOCATION WAS USED FOR ALL TERMS.)
      ABCL=1.59/QC VA
      IF(ALTSG.LE.11.0) ABCL=0.0
      ABSB=ABS(SIN(BE VA))
      CDMON=5.19/VT VA
C CUSHION RETRACTED INCREMENTS OGE FANS CLOSED
      RETCD=.004
      RETCY=0.0
      RETCM=-.004*(ABZF-7.9*ALRAD)/C  OL
      RETCLL=0.0
      RETCN=0.0
C CUSHION EXTENDED INCREMENTS FAN CLOSED
      EXTCD=.02
      EXTCY=-.05*ABSB
      EXTCM=-.02*(ABZC-5.54*ALRAD)/C  OL
      EXTCLL=-EXTCY*ABZC/B  DL
      EXTCN=EXTCY*5.54/B  DL
C CUSHION EXTENDED INCREMENTS WITH FANS OPERATIONAL
      EMCD=CDMON+ABCL*ALRAD
      EMCY=-CDMON
      EMCM=-CDMON*(ABZF-5.6*ALRAD)/C  OL+ABCL*5.54/C  OL
      EMCLL=-EMCY*ABZF/B  DL
      EMCN=EMCY*5.6/B  DL
C * * * * * * * * * * * * * * * * * * * * * * * * *
C .
C AERODYNAMIC LIFT DERIVATIVES AND COEFFICIENTS
C
C********** BASIC LIFT COEFFICIENT **********
      Z1=(.35+2.75*FLPRAD-.616*FLP2+(.15+1.865*FLPRAD
     1-.821*FLP2)*CTTOT+(5.27+.72*CTTOT)*ALRAD)
     2*COS2BE
      ZO OL=-Z1
C********** ELEVATOR DERIVATIVE **********
```

Figure 157 EASY XC-8A ACLS Model File For The Taxi Simulation (Continued)

```
      ZDEOL=-.566*COS2BE
C********** SPOILER DERIVATIVE **********
      ZSPOL=-(-.25-.573*FLPRAD)
      SPOOL=SPOIL/RPD
C        SPOIL IS DIVIDED BY RPD TO NEGATE THE EFFECTS
C        OF THE AUTOMATIC DEGREE TO RADIAN CONVERSION.
C********** TRUNK COEFFICIENT **********
      ZTROL=-TRUNK*ABCL
C * * * * * * * * * * * * * * * * * * * * * * * * *
C
C AERODYNAMIC DRAG DERIVATIVES AND COEFFICIENTS
C
C********** BASIC DRAG COEFFICIENT **********
      CL2=(ZO OL+ZDEOL*ELEOL*RPD+ZTROL+ZSPOL*SPOOL*RPD)**2.
      X1=(.026+.0394*CL2+(.115+ 46*CTTOT-.12*CTT2
     1)*FLPRAD)*COS2BE
      IF(X1.LT.0.0) X1=0.0
C          MODIFICATION OF BASIC DRAG COEFFICIENT
C          WHEN VT VA IS LESS THAN 70.0 FPS.
      X2=0.0
      IF(VT VA.GT.70.0) GO TO 15
      X2=(.229*FLPRAD+.176*FLP2)*CTTOT
      B=1.0-(VT VA-50.0)/20.0
      IF(50.0.LE.VT VA.AND.VT VA.LE.70.0) X2=X2*B
   15 CONTINUE
      XO OL=-X1-X2
C********** TRUNK COEFFICIENT **********
      XTROL=-(RETCD+TRUNK*(EXTCD+EMCD))
C********** SPOILER DERIVATIVE **********
      XSPOL=(-.0125-(.1*FLPRAD))
C * * * * * * * * * * * * * * * * * * * * * * * * *
C
C AERODYNAMIC SIDEFORCE DERIVATIVES AND COEFFICIENTS
C
C********** BETA DERIVATIVE **********
      IF(BEABS.GT..436) GO TO 18
      A=-.8-.515*FLPRAD
      B=-.1-.702*FLPRAD
      C=-.57-1.89*FLPRAD+.82*FLP2
      D=.573-1.64*FLPRAD
      IF(FLPRAD.GE..349) D=-1.145
      Y1=A+B*CTTOT+(C+D*CTTOT)*ALRAD
      GO TO 19
   18 A=.08+.225*FLPRAD
      B=(.025+.322*FLPRAD)*CTTOT-.079*BERAD
      IF(ABS(CTTOT).LT..1) B=B+.079*BERAD
      IF(BERAD-PIH) 500,501,501
  500 Y1=-(1.0/BEABS)*(1.38*SIN2BE+A*(1.0
     1-(BEABS-.436)/1.134)+B)
```

Figure 157 EASY XC-8A ACLS Model File For The Taxi Simulation (Continued)

```
      GO TO 19
 501  Y1=-(1.0/BEABS)*(1.38*SIN2BE+A*(PIH-BEABS)/
     1   1.134+B)
  19  YB DL=Y1
C********** AILERION DERIVATIVE **********
      YDADL=.0277*(.0473+.0545*FLPRAD)*FFBETA
C********** RUDDER DERIVATIVE AND EFFECTIVENESS **********
      YDRDL=.407
      YBRDL=FFBETA
C********** ROLL RATE DERIVATIVE **********
      YP DL=-.06
C********** YAW RATE DERIVATIVE **********
      YR DL=.704
C********** TRUNK COEFFICIENT **********
      YTRDL=RETCY+TRUNK*(EXTCY+EMCY)
C * * * * * * * * * * * * * * * * * * * * * * * * *
C
C AERODYNAMIC PITCHING MOMENT DERIVATIVES AND COEFFICIENTS
C
C********** BASIC PITCHING MOMENT COEFFICIENT **********
      IF(CTTOT.GE.1.0) GO TO 20
      A=.2-.2*CTTOT
      B=.251-.036*CTTOT
      C=-.245*CTTOT
      D=-3.69+3.616*CTTOT
      E=-1.645-5.375*CTTOT
      F=4.71*CTTOT
      GO TO 21
  20  A=.2*(CTTOT-1.0)
      B=.215+1.002*(1.0-CTTOT)
      C=-.49+.245*CTTOT
      D=-.074+1.566*(1.0-CTTOT)
      E=-14.04+7.02*CTTOT
      F=9.42-4.71*CTTOT
  21  M1=A+B*FLPRAD+C*FLP2+(D+E*FLPRAD+F*FLP2)
     1*(ALRAD+.1745)**2.
      MO OL=M1
C********** PITCH RATE DERIVATIVE **********
      MQ OL=-38.4
C********** ALPHA DOT DERIVATIVE **********
      MADOL=-7.-5.*CTTOT
      IF(CTTOT.GE.1.0) MADOL=-12.0
C********** TRUNK COEFFICIENT **********
      MTROL=(RETCM+EXTCM+EMCM)*TRUNK
C********** ELEVATOR DERIVATIVE **********
      MDEOL=-2.43*COS2BE
C********** SPOILER DERIVATIVE
      MSPOL=(.02+.1145*FLPRAD)
C * * * * * * * * * * * * * * * * * * * * * * * * *
```

Figure 157 EASY XC-8A ACLS Model File For The Taxi Simulation (Continued)

```
C
C AERODYNAMIC ROLL MOMENT DERIVATIVES AND COEFFICIENTS
C
C********** BETA DERIVATIVE **********
      IF(BEABS.GT..436) GO TO 31
      L1=(-.164+.043*FLPRAD+(.014+.1575*FLPRAD-.0924*FLP2)
     1*CTTOT+(.172+(.1035-.0186*FLPRAD-.082*FLP2)
     2*CTTOT)*ALRAD)
      GO TO 32
 31   A=.01875*FLPRAD
      B=(.0041+.0326*FLPRAD)*CTTOT
      IF(BEABS-PIH) 510,511,511
 510  L1=(1.0/BEABS)*(-.022-.235*SIN2BE*COS2BE+
     1A*(1.0-(BEABS-.436)/1.134)+B)
      GO TO 32
 511  L1=(1.0/BEABS)*(-.022-.235*SIN2BE*COS2BE+A
     1   *(PIH-BEABS)/1.134+B)
 32   LB DL=L1
C********** ROLL RATE DERIVATIVE **********
      LP DL=-.53-.1*FLPRAD+(.08-.0572*FLPRAD)*CTTOT
     1+.516*ALRAD
C**********YAW RATE DERIVATIVE **********
      LR DL=.15+.587*FLPRAD+.975*ALRAD
C********** AILERION DERIVATIVE **********
      LDADL=(.0473+.0545*FLPRAD)*FFBETA
C********** RUDDER DERIVATIVE AND EFFECTIVENESS PARAMETER **********
      LDRDL=.184*SIN(.265-ALRAD)
      LBRDL=FFBETA
C********** TRUNK COEFFICIENT **********
      LTRDL=RETCLL+TRUNK*(EXTCLL+EMCLL)
C * * * * * * * * * * * * * * * * * * * * * * * * *
C
C AERODYNAMIC YAW MOMENTS DERIVATIVES AND COEFFICIENTS
C
C********** BETA DERIVATIVE **********
      IF(BEABS.GT..436) GO TO 37
      A=.125+.0329*FLPRAD
      B=-.015-.0415*FLPRAD
      C=.057+.52*FLPRAD-.495*FLP2
      D=-.0573-.181*FLPRAD
      IF(FLPRAD.GE..349) D=-.1204+.1315*(FLPRAD-.349)
      N1=A+B*CTTOT+(C+D*CTTOT)*ALRAD
      GO TO 38
 37   A=.02475+.01435*FLPRAD
      B=(.006+.0236*FLPRAD)*CTTOT
      IF(BEABS-PIH) 520,521,521
 520  N1=(1.0/BEABS)*(.106*SIN2BE+A*(1.0
     1-(BEABS-.436)/1.134)-B)
      GO TO 38
```

Figure 157 EASY XC-8A ACLS Model File For The Taxi Simulation (Continued)

```
521  N1=(1.0/BEABS)*(.106*SIN2BE+A*(PIH-BEABS)/1.134-B)
 38  NB DL=N1
C********** YAW RATE DERIVATIVE **********
      NR DL=-.22-.06*FLPRAD-(.129+.308*FLPRAD)*ALRAD
C********** ROLL RATE DERIVATIVE **********
      NP DL=((-.1+.009*CTTOT)*FLPRAD)/.699+
     1(-.5-.145*CTTOT*(FLPRAD-.699)/.699)*ALRAD
C********** AILERION DERIVATIVE **********
      NDADL=-(.0467+.704*ALRAD)*LDADL
C********** RUDDER DERIVATIVE AND EFFECTIVENESS PARAMETER **********
      NDRDL=-.184*COS(.265-ALRAD)
      NBRDL=FFBETA
C********** TRUNK COEFFICIENT **********
      NTRDL=RETCN+TRUNK*(EXTCN+EMCN)
C XP1OL CORRECTS THE AERODYNAMIC DATA FOR A C.G. LOCATION
C DIFFERENT FROM 40 PERCENT MAC USED IN DERIVING THE DATA.
      XP1OL=C  OL*(.01*CGRAV-.4)
C*****************************************************************
C*****************************************************************
C  THE FOLLOWING FORTRAN STATEMENTS EVALUATE THE GROUND EFFECTS
C  FOR THE AERODYNAMIC MODEL. THE PROCEEDURE USED TO EVALUATE
C  GROUND EFFECTS WAS TAKEN FROM DATCOM SECTION 4.7 METHOD 1.
C  THE REQUIRED INPUT PARAMETERS ARE
C*********ROOTC=ROOT CHORD (CR) LENGTH (FEET)
C*********SH=HORIZONTAL STABILIZER AREA (SQUARE FEET)
C*********ASPECT=ASPECT RATIO
C*********FS25CR,WL25CR=THE FUSELAGE STATION AND WATER
C         LINE LOCATIONS OF .25 ROOT CHORD.
C*********FS75CR,WL75CR=THE FUSELAGE STATION AND WATER
C         LINE LOCATIONS OF THE .75 ROOT CHORD.
C*********FSTAIL,WLTAIL=FUSELAGE STATION AND WATER
C         LINE LOCATIONS OF THE HORIZONTAL STABILIZERS
C         AERODYNAMIC CENTER.
C  THE REQUIRED INPUT TABLES ARE
C*********TBCL0,TBCL20,TBCL40= THE TOTAL AIRPLANES COEFFICIENT
C         OF LIFT AT 0.0,20.0,40.0 DEG. FLAPS RESPECTIVELY.
C         THESE TABLES ARE FUNCTIONS OF ALPHA AND CTTOT.
C*********TBCLH=THE HORIZONTAL STABILIZER COEFFICIENTS OF
C         INPUT AS FUNCTIONS OF HORIZONTAL ALPHA AND ELEOL.
C*********TBX=THE X VARIABLES FROM DATCOM FIGURE 4.7.1-14.
C*********TBLLO=THE L/LO-1. VARIABLE FROM DATCOM FIGURE 4.7.1-15.
C*********TBDEL=THE DELTA DELTA CL VARIABLE FROM DATCOM
C         FIGURE 4.7.1-17 FOR SLOTTED FLAPS.
C  ASSUMPTIONS
C         1. SWEEP ANGLE OF THE QUARTER CHORD LINE
C            OF THE WING = 0.0 DEG..
C         2. THE DYNAMIC PRESSURE OF THE HORIZONTAL
C            STABILIZER EQUALS THE DYNAMIC PRESSURE
C            AT INFINITY.
```

Figure 157 EASY XC-8A ACLS Model File For The Taxi Simulation (Continued)

```
C         3. THE WINGS ARE STRAIGHT AND PERPENDICULAR
C            TO THE X-Z PLANE. THUS H.75CR=H.75B/2.
C            SEE DATCOM FIGURE 4.7.1-14.
C         4. THE MAIN WINGS AERODYNAMIC CENTER IS ASSUMED
C            TO BE AT .25C..
C  IF THE AIRCRAFTS ALTITUDE IS GREATER THAN TWICE THE
C  WING SPAN THE GROUND EFFECTS ARE SKIPPED.
      DELCLGR=0.0
      DELCDLG=0.0
      DELCMG=0.0
      IF(ALTSG.GT.2.*B  DL) GO TO 777
C  THE NEXT GROUPING OF FORTRAN STATEMENTS EVALUATES
C  FREQUENTLY USED VARIABLES IN THE GROUND EFFECTS
C  CALCULATIONS.
C
      SRATIO=SH/S  VA
      EPSILON=AL VA*(.24-.0015*FLAPS)
      TAILA=AL VA-EPSILON
      X75CR=(FSCG-FS75CR)/12.0
      Z75CR=(WLCG-WL75CR)/12.0
      X25CR=(FSCG-FS25CR)/12.0
      Z25CR=(WLCG-WL25CR)/12.0
      XTAIL=(FSCG-FSTAIL)/12.0
      ZTAIL=(WLCG-WLTAIL)/12.0
      RPD=.01745329
      SP=SIN(PITSG*RPD)
      CR=COS(ROLSG*RPD)
      CP=COS(PITSG*RPD)
      CRCP=CR*CP
      H75CR=X75CR*SP-Z75CR*CRCP+ALTSG
      IF(H75CR.LE.11.25) GO TO 777
      H25CR=X25CR*SP-Z25CR*CRCP+ALTSG
      HTAIL=XTAIL*SP-ZTAIL*CRCP+ALTSG
      H75=2.*H75CR/B  DL
      H25=H25CR/ROOTC
      CRB=ROOTC/B  DL
C  THE NEXT GROUPING OF FORTRAN STATEMENTS EVALUATES
C  THE TOTAL AIRCRAFTS COEFFICIENT OF LIFT BY TABLE
C  LOOK UP ROUTINES INPUT AS FUNCTIONS OF ALPHA AND CTTOT.
C  THERE ARE 3 INPUT TABLES FOR FLAP SETTINGS OF 0.0,20.0
C  AND 40.0 DEGREES. A LINEAR INTERPELATION IS MADE
C  FOR FLAP SETTINGS BETWEEN THESE VALUES.
      I=0
      IF(FLAPS.GE.20.) GO TO 2
      T=FLAPS/20.0
      GO TO 4
    2 T=(FLAPS-20.)/20.
    4 A=AL VA
    5 I=I+1
```

Figure 157 EASY XC-8A ACLS Model File For The Taxi Simulation (Continued)

```
      IF(FLAPS.GE.20.) GO TO 14
      COEL1=TBLU2(A,CTTOT,TBCL0(7),TBCL0(4),TBCL0(17)
     1,1,1,10,3,10,3)
      COEL2=TBLU2(A,CTTOT,TBCL20(7),TBCL20(4),TBCL20(17)
     1,1,1,10,3,10,3)
      GO TO 150
   14 COEL1=TBLU2(A,CTTOT,TBCL20(7),TBCL20(4),TBCL20(17)
     1,1,1,10,3,10,3)
      COEL2=TBLU2(A,CTTOT,TBCL40(7),TBCL40(4),TBCL40(17)
     1,1,1,10,3,10,3)
  150 CL=(COEL2-COEL1)*T+COEL1
      GO TO (10,200,30,40) I
C  CALCULATION OF THE GROUND EFFECTS UPDATE FOR
C  THE LIFT COEFFICIENT.
   10 CLWBH=CL
      CLH=TBLU2(TAILA,ELEOL,TBCLH(7),TBCLH(4),TBCLH(15)
     1,1,1,8,3,8,3)
C*********CLFWB=IS THE WING-BODY LIFT COEFFICIENT INCLUDING
C         FLAP EFFECTS, OUT OF GROUND EFFECT.
      CLFWB=CLWBH-CLH*SRATIO
      X=TBLU1(H75,TBX(4),TBX(15),1,11)
      IF(X.LT.0.0) X=0.0
      ALUP=AL VA+.5
      ALDN=AL VA-.5
      EUP=ALUP*(.24-.0015*FLAPS)
      EDN=ALDN*(.24-.0015*FLAPS)
      AHUP=ALUP-EUP
      AHDN=ALDN-EDN
      A=ALUP
      GO TO 5
  200 CLUP=CL
      A=ALDN
      GO TO 5
   30 CLDN=CL
      CLHUP=TBLU2(AHUP,ELEOL,TBCLH(7),TBCLH(4),TBCLH(15)
     1,1,1,8,3,8,3)
      CLHDN=TBLU2(AHDN,ELEOL,TBCLH(7),TBCLH(4),TBCLH(15)
     1,1,1,8,3,8,3)
C*********CLAWB=IS THE WING-BODY LIFT CURVE SLOPE,PER DEGREE
C         OUT OF GROUND EFFECT. IT IS EVALUATED BY CALCULATING
C         CLFWB FOR AL VA+.5 AND AL VA-.5 DEGREES. CLAWB IS
C         ASSIGNED THE AVERAGE SLOPE CALCULATED FOR THIS
C         ONE DEGREE CHANGE IN ALPHA.
      CLAWB=CLUP-CLDN-SRATIO*(CLHUP-CLHDN)
      CKCL=CLWBH*9.1196
      OLL=TBLU2(H25,CKCL,TBLLO(7),TBLLO(4),TBLLO(19)
     1,1,1,-12,3,12,3)
      R=SQRT(1.+H75**2.)-H75
      DELDEL=TBLU1(H25,TBDEL(4),TBDEL(9),1,5)
```

Figure 157 EASY XC-8A ACLS Model File For The Taxi Simulation (Continued)

```
      IF(DELDEL.GT.0.0) DELDEL=0.0
      CK11=9.12/ASPECT+7.16*CRB
      CK12=ASPECT*CRB/2.
      CK13=(FLAPS/50.)**2.
C*********DELAL=THE CHANGE IN ALPHA AT THE PRESENT
C          ALTITUDE.
      DELAL=-CK11*CLFWB*X-(CK12/CLAWB)*OLL*CLFWB*R-CK13*DELDEL/CLAWB
      IF(DELAL.GT.0.0) DELAL=0.0
      ALGR=AL VA-DELAL
      A=ALGR
      GO TO 5
   40 CL1=CL
      DELCLGR=CLWBH-CL1
      ZO OL=ZO OL+DELCLGR
C  CALCULATION OF THE GROUND EFFECT UPDATE FOR
C  THE MOMENT M COEFFICIENT.
      FLAPNO=TBLU2(AL VA,CTTOT,TBCLO(7),TBCLO(4),TBCLO(17)
     1,1,1,10,3,10,3)
C*********CLWB=THE WING-BODY LIFT COEFFICIENT, FLAPS RETRACTED,
C          OUT OF GROUND EFFECT.
      CLWB=FLAPNO-SRATIO*CLH
C*********DELCLF=THE CHANGE IN LIFT COEFFICIENT DUE TO FLAPS,
C          OUT OF GROUND EFFECT.
      DELCLF=CLWBH-FLAPNO
C*********BEFF=THE EFFECTIVE WING SPAN.
      BEFF=(CLWB+DELCLF)/((CLWB/74.4)+(DELCLF/58.8))
      BEFF2=BEFF**2.
      SN1=BEFF2+4.*(HTAIL-H25CR)**2.
      SN2=BEFF2+4.*(HTAIL+H25CR)**2.
      DELE=EPSILON*SN1/SN2
      HAL=TAILA+DELE
      CLHGR=TBLU2(HAL,ELEOL,TBCLH(7),TBCLH(4),TBCLH(15)
     1,1,1,8,3,8,3)
      DELCLHG=CLHGR-CLH
C*********DELCMHG=THE CHANGE IN PITCHING MOMENT COEFFICIENT
C          PRODUCED BY THE HORIZONTAL STABILIZER. A .4MAC
C          C.G. LOCATION WAS ASSUMED.
      DELCMHG=-DELCLHG*SRATIO*4.304
      DCLWBG=DELCLGR+DELCLHG*SRATIO
C*********DCMUBG=THE CHANGE IN PITCHING MOMENT FOR THE
C          WING. A .4C C.G. LOCATION WAS ASSUMED.
      DCMWBG=-.15*DCLWBG
      DELCMG=DCMWBG+DELCMHG
      MO OL=MO OL+DELCMG
C CALCULATION OF THE GROUND EFFECTS UPDATE FOR
C  THE DRAG COEFFICIENT.
      SIGMA=EXP(-2.48*H75**.768)
      DELCDLG=SIGMA*(CLFWB**2.)/(ASPECT*3.14159)
      XO OL=XO OL+DELCDLG
```

Figure 157 EASY XC-8A ACLS Model File For The Taxi Simulation (Continued)

```
 777  CONTINUE
C**********************************************************************
C**********************************************************************
LOCATION=80     FL      INPUTS=SG,VA(MAC=AMN)
LOCATION=76     FN      INPUTS=FL,TS(PT=P,2)
FORTRAN STATEMENTS
C   THE FOLLOWING STATEMENT CONVERTS THE LBS/SEC FLOW RATE TO
C   LBS/MIN. THIS VALUE IS THEN DOUBLED TO ACCOUNT FOR THE TWO
C   FANS IN THE SYSTEM.  (I.E. 60*2=120)
      WTRTS=120.*W2 FN
      EPCTS=0.0
      IF(IERR.NE.1) GO TO 82
      ITIME=TIME*10.0
      PRNT=(TIME*10.0)-FLOAT(ITIME)
      IF(PRNT.EQ.0.0) EPCTS=1.0
  82  CONTINUE
LOCATION=17     TS      INPUTS=SG,FL(PAM=PA),FN(T,2=TTR)
LOCATION=14     S2      INPUTS=
TS(FXT=FX,2,FYT=FY,2,FZT=FZ,2,TXT=TX,2,TYT=TY,2,TZT=TZ,2)
LOCATION=34     OL      INPUTS=VA,S2
LOCATION= 1     DL      INPUTS=VA,OL,S2
LOCATION=40     SG      INPUTS=OL,DL
FORTRAN STATEMENTS
      RPD=.01745329
      SP=SIN(PITSG*RPD)
      SR=SIN(ROLSG*RPD)
      CP=COS(PITSG*RPD)
      CR=COS(ROLSG*RPD)
      ACELCG=(FX2OL*SP-FY2DL*SR*CP-FZ2OL*CR*CP)/(32.174*MA1OL)-1.
      PCUSH=(PC TS-PAMFL)*144.0
      PTRUNK=(PT TS-PAMFL)*144.0
C*********ACELCG=THE VERTICAL ACCELERATION OF THE C.G.. (G)
C*********PCUSH=GAGE CUSHIOH PRESSURE (PSFG)
C*********PTRUNK=GAGE TRUNK PRESSURE (PSFG)
END OF MODEL
PRINT
```

Figure 157 EASY XC-8A ACLS Model Tile For The Taxi Simulation (Concluded)

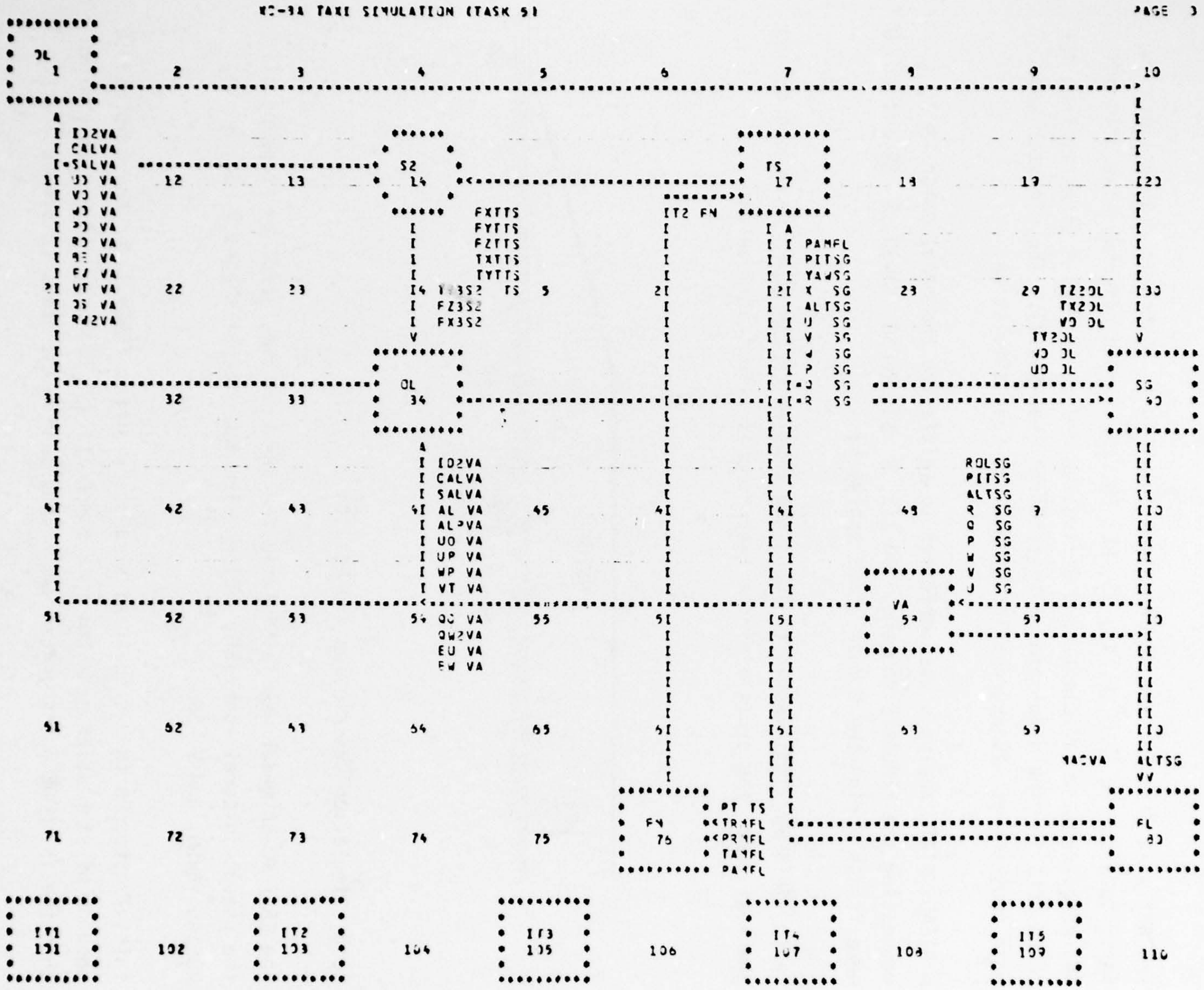


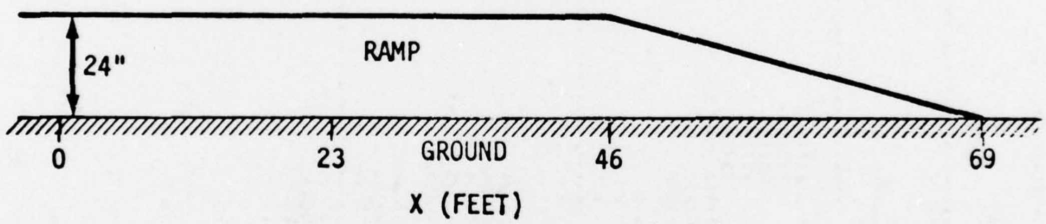
Figure 158 EASY XC-8A ACLS Model Schematic For The Taxi Simulation

11.2.2 Analysis File

The analysis file for the XC-8A taxi simulation is listed in Figure 159. The input data for OL, DL, SG, VA, TS, FL, FN, the ground effects, wing skid and the aerodynamic coefficients and derivative coding was taken directly from the XC-8A ACLS landing simulation analysis file (see 9.2.2). The elevator was held constant at -25 degrees throughout the taxi simulation.

A STEADY STATE analysis was performed to initialize the model prior to requesting the SIMULATE command. FO IT2, R SG, ROLSG, YAWSG, X SG and Y SG were frozen during the steady state analysis.

Table ZTRTS was implemented to input the ramp which the XC-8A descended during the taxi run. The profile of the test ramp is illustrated below:



11.3 Simulation Description and Results

The EASY XC-8A model was initialized for the taxi simulation by freezing all the pseudo-integral controllers along with the lateral states P SG, R SG, ROLSG, YAWSG, and V SG.

Initial attempts to run the taxi simulation with a TINC=.1 for the complete 16 seconds of simulation resulted in a computational failure. It was determined that the INT MODE 6 integrator was taking such large steps (2 seconds) after 2

```
TABLE=TBCL0,10,3
0.0,0.8,1.2
-4.0,-2.0,0.0,2.0,4.0,6.0,8.0,10.0,12.0,14.0
-.0154,.1523,.3378,.5153,.6988,.8734,1.0385
1.1809,1.3020,1.4271
-.0480,.1909,.4496,.7179,.9861,1.2423,1.4788
1.6915,1.8735,2.0650
-.0692,.2094,.4975,.7855,1.0790,1.3720,1.6264
1.8644,2.0840,2.2780
TABLE=TBCL20,10,3
0.0,0.8,1.2,
-4.0,-2.0,0.0,2.0,4.0,6.0,8.0,10.0,12.0,14.0
1.0055,1.1849,1.3435,1.4986,1.6568,1.8134
1.9626,2.0934,2.2089,2.3156
1.4464,1.7130,1.9513,2.1828,2.3979,2.6214
2.8429,3.0386,3.2056,3.3703
1.6034,1.8913,2.1564,2.3988,2.6470,2.8904
3.1287,3.3488,3.5450,3.7086
TABLE=TBCL40,10,3
0.0,0.8,1.2
-4.0,-2.0,0.0,2.0,4.0,6.0,8.0,10.0,12.0,14.0
1.6639,1.8323,1.9766,2.1411,2.3094,2.4749
2.5999,2.7305,2.8406,2.9339
2.4030,2.6580,2.8879,3.1195,3.3492,3.5917
3.7955,3.9699,4.1346,4.2591
2.6590,2.9325,3.1828,3.4399,3.7065,3.9480,4.1946
4.4070,4.5844,4.7244
TABLE=TBX,11
0.0,.1,.2,.3,.4,.5,.6,.7,.8,.9,1.0
1.0334,.7145,.5360,.4176,.3239,.2586,.2080
.1669,.1352,.1087,.0899
TABLE=TBCLH,8,3
-25.0,0.0,15.0
-18.0,-16.0,-14.0,8.0,10.0,12.0,14.0,16.0
-1.6755,-1.6289,-1.5699,-.3587,-.2409
-.1127,.00755,.1004
-1.0937,-1.0272,-.9288,.4601,.5818
.7110,.7566,.6414
-.6462,-.5369,-.4162,1.0091,1.1333,1.0773
.9537,.8311
TABLE=TBLL0,12,3
10.0,18.0,24.0
.2,.4,.6,.8,1.0,1.2,1.4,1.6,1.8,2.0,2.2,2.4
.2118,.06113,.00509,-.01720,-.02592,-.02780,-.02880,-.02938
-.02902,-.02856,-.02814,-.02824
-.05664,-.09657,-.1098,-.1063,-.09462,-.08550,-.07764,-.07006
-.06439,-.06119,-.05832,-.05740
-.1942,-.1980,-.1855,-.1648,-.1426,-.1250,-.1114
-.09891,-.08937,-.08117,-.07524,-.07178
```

Figure 159 EASY XC-8A ACLS Analysis File For The Taxi Simulation

```
TABLE=TBDEL,5
.2,.4,.6,.8,1.0
-.09898,-.06246,-.03709,-.01953,-.005234
TABLE=ABLTS,9
71.9,0.0,71.9,.05,0.0,.5
47.1,10.0,57.1,.12,0.0,.5
44.2,15.0,56.8,.17,0.0,.3
TABLE=XYZTS,32
148.91,1.6,62.5,80.0
147.81,4.5,62.5,60.0
145.81,6.9,62.5,40.0
142.41,8.7,62.5,15.0
124.56,9.0,62.5,0.0
93.56,9.0,62.5,0.0
62.56,9.0,62.5,0.0
29.41,9.0,62.5,0.0
5.91,9.0,62.5,0.0
-17.84,9.0,62.5,0.0
-41.34,9.0,62.5,0.0
-65.09,9.0,62.5,0.0
-83.79,8.8,62.5,-12.5
-87.09,7.4,62.5,-35.0
-89.49,4.8,62.5,-57.5
-90.79,1.6,62.5,-80.0
TABLE=DM TS,16
20.0,.097,20.0,.097,20.0,.097,30.0,.097
31.0,.097,31.0,.097,31.0,.097,34.0,.097
13.25,.097,34.0,.097,13.25,.097,34.0,.097
22.5,.097,22.5,.097,25.0,.097,20.0,.097
TABLE=IALTS,32
1,.02189,21.0,23.0
1,.02189,21.0,23.0
2,.01911,22.5,14.0
2,.01911,22.5,14.0
3,.01911,22.25,14.0
3,.01911,22.25,14.0
3,.01911,22.25,14.0
3,.0025,19.5,21.0
3,.01911,23.5,13.0
3,.0025,19.5,21.0
3,.01911,23.5,13.0
3,.0025,19.5,21.0
2,.01911,22.5,14.0
2,.01911,22.5,14.0
1,.02189,21.0,23.0
1,.02189,21.0,23.0
TABLE=ENDTS,2
9.0,.45,9.0,.45
TABLE=SPHTS,8,3
```

Figure 159 EASY XC-8A ACLS Analysis File For The Taxi Simulation (Continued)

```
1,2,3
0.0,33.,66.,100.,150.,200.,300.,500.
0.0,.2288,.6903,1.0079,1.1108,1.1963,1.3127,1.4880
0.0,.7809,1.8073,2.1367,2.2924,2.4125,2.5653,2.76
0.0,.9804,2.2110,2.5446,2.7195,2.8521,3.0180,3.2197
TABLE=STHTS,8,3
1,2,3
0.0,33.,66.,100.,150.,200.,300.,500.
0.0,.9804,2.2110,2.5446,2.7195,2.8521,3.0180,3.2197
0.0,.9804,2.2110,2.5446,2.7195,2.8521,3.0180,3.2197
0.0,.0135,.02702,.04094,.0614,.08187,.1228,.16374
TABLE=PM TS,14
7,0.0,0.0,.9
8,.8,5.0,.6
9,0.0,0.0,.8
10,.8,5.0,.6
11,0.0,0.0,.8
12,.8,5.0,.6
13,0.0,0.0,.9
TABLE=RELTS,2
0.0,50.0
0.0,0.0
TABLE=BWTTS,4
344.1,160.0,344.1,160.0
2,4,23,1
TABLE=FANFN,16,3
5260.,5880.,6370.
1.0,1.051,1.075,1.100,1.126,1.150,1.175,1.195
1.201,1.221,1.242,1.251,1.279,1.289,1.300,1.339
62.56,60.64,59.11,57.0,54.14,50.66,45.82,39.37
-.203,-1.444,-2.615,-3.158,-4.715,-5.258,-5.897,-8.153
69.18,67.8,66.81,65.22,63.39,61.3,58.74,56.25,55.48
51.73,45.52,-.275,-1.939,-2.483,-3.05,-5.353
74.87,73.67,72.99,72.09,70.92,69.55,67.97,66.25
65.77,63.56,60.81,59.16,50.28,-.231,-.763,-2.923
TABLE=STAFN,17
1.0,1.021,1.041,1.061,1.081,1.100,1.120,1.140
1.16,1.181,1.201,1.222,1.241,1.260,1.281,1.30,1.312
0.0,7.0,12.54,17.19,21.23,25.01,28.45,31.54
34.57,37.42,40.05,42.95,45.31,47.82,50.36,52.82,54.17
TABLE,ZTRTS,4
24.0,24.0,24.0,0.0
0.0,0.0,0.0,0.0
PARAMETER VALUES
NUIFN=.2,NUFFN=.87
CORFN=1.0,RPMFN=5260.
FSENG=197.0,BLENG=183.0,WLENG=191.0
FSCG=343.41,BLCG=0.0,WLCG=160.0
FS75CR=406.81,WL75CR=206.06,FS25CR=336.79
```

Figure 159 EASY XC-8A ACLS Analysis File For The Taxi Simulation (Continued)

```
WL25CR=206.06,FSTAIL=889.65,WLTAIL=403.84
ROOTC=11.67,ASPECT=9.75,SH=233.0
THETA=0.0,TRUNK=1.0
CGRAV=28.0,SPOIL=0.0,FLAPS=7.0
ID1VA=3,VS VA=11.822,ALSVA=1.2
S  VA=945.0,IDGVA=6
WCUTS=0.0,TCUTS=560.0
ANETS=-16,CDGTS=1.0,CD1TS=.62
CD2TS=.2,TAUTS=.005,DMPTS=.02
VU TS=6.0,PTMTS=3.0
SPBTS=0.0
MA1OL=1087.83,C   OL=10.29,ISWOL=3
B  DL=96.0
IXXSG=183700.,IYYSG=228200.
IZZSG=365100.,IXZSG=29220.
FSSKID=340.0,BLSKID=432.5,WLSKID=90.0
SPLOW=2666.,SPHIGH=50000.,DEFMX=1.875
SKIDMU=.02,SKID=1.0
GKIIT1=10.0,GKIIT2=10.0,GKIIT3=10.0
GKIIT4=10.0,GKIIT5=10.0
INITIAL CONDITIONS
U  SG=11.822,V  SG=0.0,W  SG=0.0
P  SG=0.0,Q  SG=0.0,R  SG=0.0
ROLSG=0.0,PITSG=0.0,YAWSG=0.0
ALTSG=10.0,X  SG=0.0,Y  SG=0.0
PT TS=16.627,VT TS=850.29
PC TS=14.731,VC TS=139.47
FO IT1=130.,FO IT2=-25.0
FO IT3=1.0,FO IT4=0.0,FO IT5=119.
ERROR CONTROLS
U  SG=.001,V  SG=.0001,W  SG=.001
P  SG=.0001,Q  SG=.0001,R  SG=.0001
ROLSG=.0001,PITSG=.0001,YAWSG=.0001
ALTSG=.0001,X  SG=.0001,Y  SG=.0001
PT TS=.0001,VT TS=.001
PC TS=.0001,VC TS=.001
FO IT1=.0001,FO IT2=.0001,FO IT3=.0001
FO IT4=.0001,FO IT5=.0001
PRINT CONTROL=4
SS ITERATIONS=60
ALL STATES
INT CONTROLS,R  SG=0,ROLSG=0,YAWSG=0
FO IT2=0,X  SG=0,Y  SG=0
LINEAR ANALYSIS
STEADY STATE
XIC-X
LINEAR ANALYSIS
TINC=.2,TMAX=2.0,INT MODE=6
ALL STATES
```

Figure 159 EASY XC-8A ACLS Analysis File For The Taxi Simulation (Continued)

```
INT CONTROLS
FO IT1=0,FO IT2=0,FO IT3=0
FO IT4=0,FO IT5=0
ROLSG=0,YAWSG=0,V  SG=0
P  SG=0,R  SG=0
LINEAR ANALYSIS
PRINTER PLOTS,PLOT ON
DISPLAY1
PCUSH,VS,TIME
PTRUNK,VS,TIME
VC TS,VS,TIME
VT TS,VS,TIME
DISPLAY2
ROLSG,VS,TIME
PITSG,VS,TIME
YAWSG,VS,TIME
Q  SG,VS,TIME
DISPLAY3
X  SG,VS,TIME
Y  SG,VS,TIME
ALTSG,VS,TIME
ACELCG,VS,TIME
VKNOTS,VS,TIME
TITLE     XC-8A TAXI SIMULATION
PLOT ID     PAUL R. PERKINS  MS 47-03
SIMULATE
XIC-X
INITIAL TIME=2.0
TINC=.01,TMAX=7.0,OUTRATE=20.
SIMULATE
XIC-X
INITIAL TIME=7.0
TINC=.2,TMAX=16.0,OUTRATE=1
SIMULATE
XIC-X
LINEAR ANALYSIS
```

Figure 159 EASY XC-8A ACLS Analysis File For The Taxi Simulation (Concluded)

seconds of integration time that when it detected the down ramp and decreased the step size a negative trunk gauge pressure was calculated. This negative trunk pressure resulted in a negative pressure ratio calculation in component FN which in turn was raised to a fractional exponent creating a fatal error terminating the program execution before the integrator could recover from the large time step increments. To prevent the INT MODE 6 integrator from taking excessively large time steps, the simulation was broken down into three simulation time segments with a smaller TINC when the aircraft encounters the ramp. These simulation time increments and respective TINC values are listed below:

Simulation Time (sec)	TINC
0 - 2	0.1
2 - 7	0.01
7 - 16	0.1

11.4 Discussion

Figure 160-a, -b, -c and -d are EASY XC-8A taxi simulation results for aircraft pitch, altitude, airspeed and cushion pressure. Pitch was the only flight test parameter plotted for comparison with the EASY simulation results because the random error in this flight test data was greater than the changes experienced as the aircraft went down the ramp. The EASY simulation for pitch demonstrated the same trends which were present in the XC-8A flight test data. No trunk instability was noticed in the EASY simulation results and the oscillations produced as the aircraft descended the ramp were stable and decayed in several cycles.

11.5 Conclusions

Based on the qualitative correlation between the XC-8A aircraft and the EASY simulation during taxi the EASY XC-8A ACLS model is valid for taxi maneuvers.

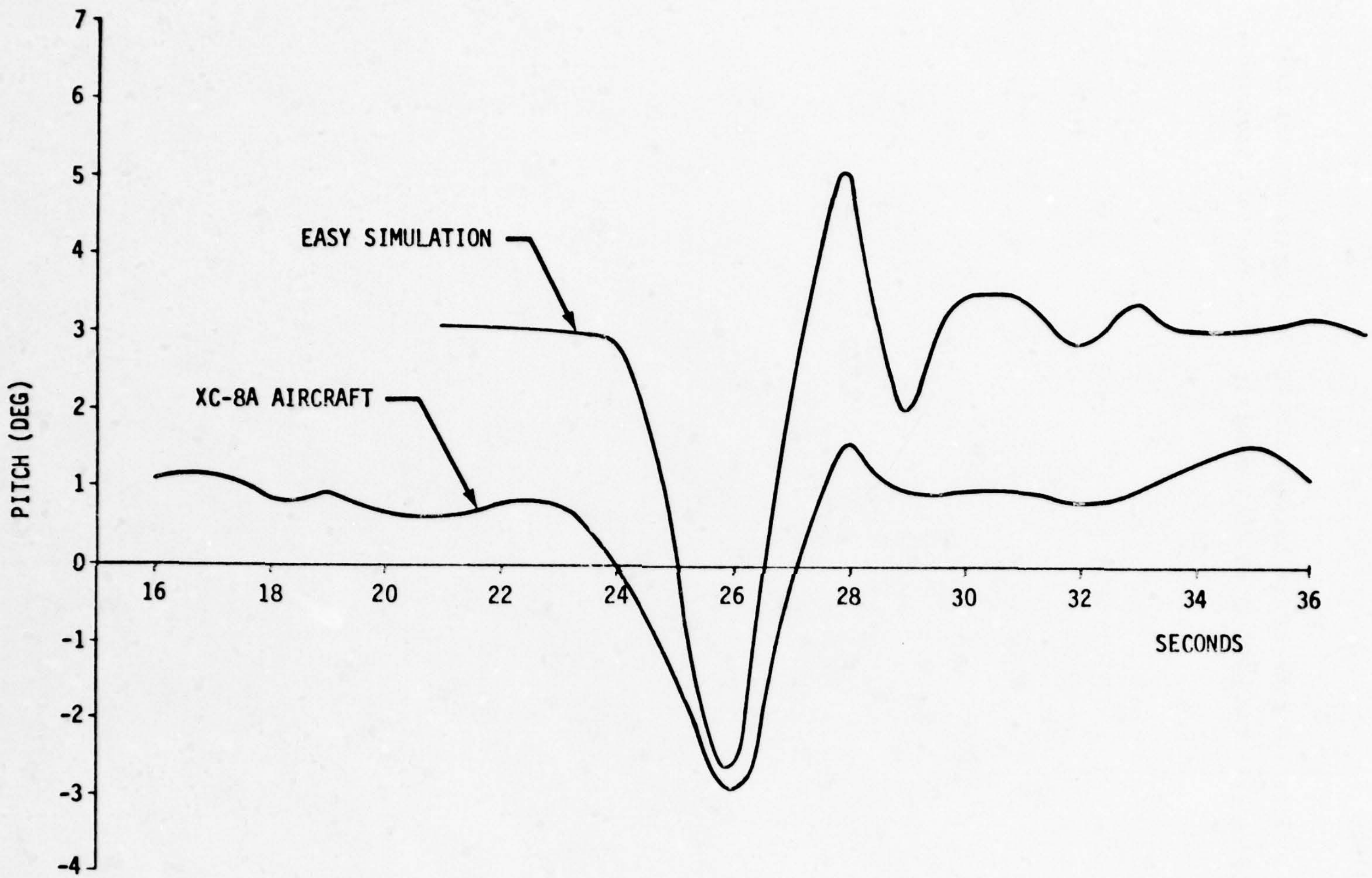


Figure 160 (a) EASY XC-8A Taxi Simulation Results

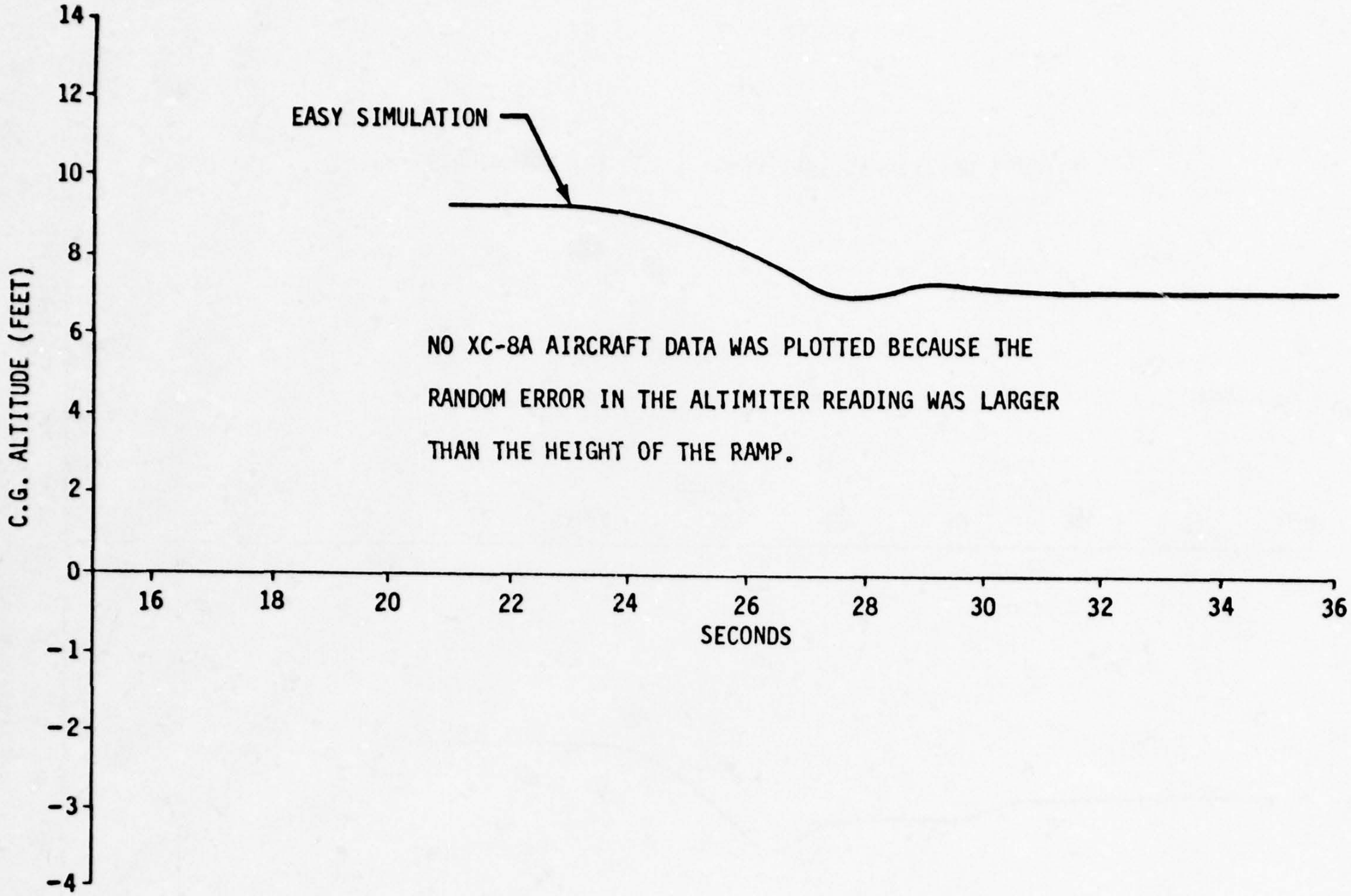


Figure 160 (b) EASY XC-8A Taxi Simulation Results

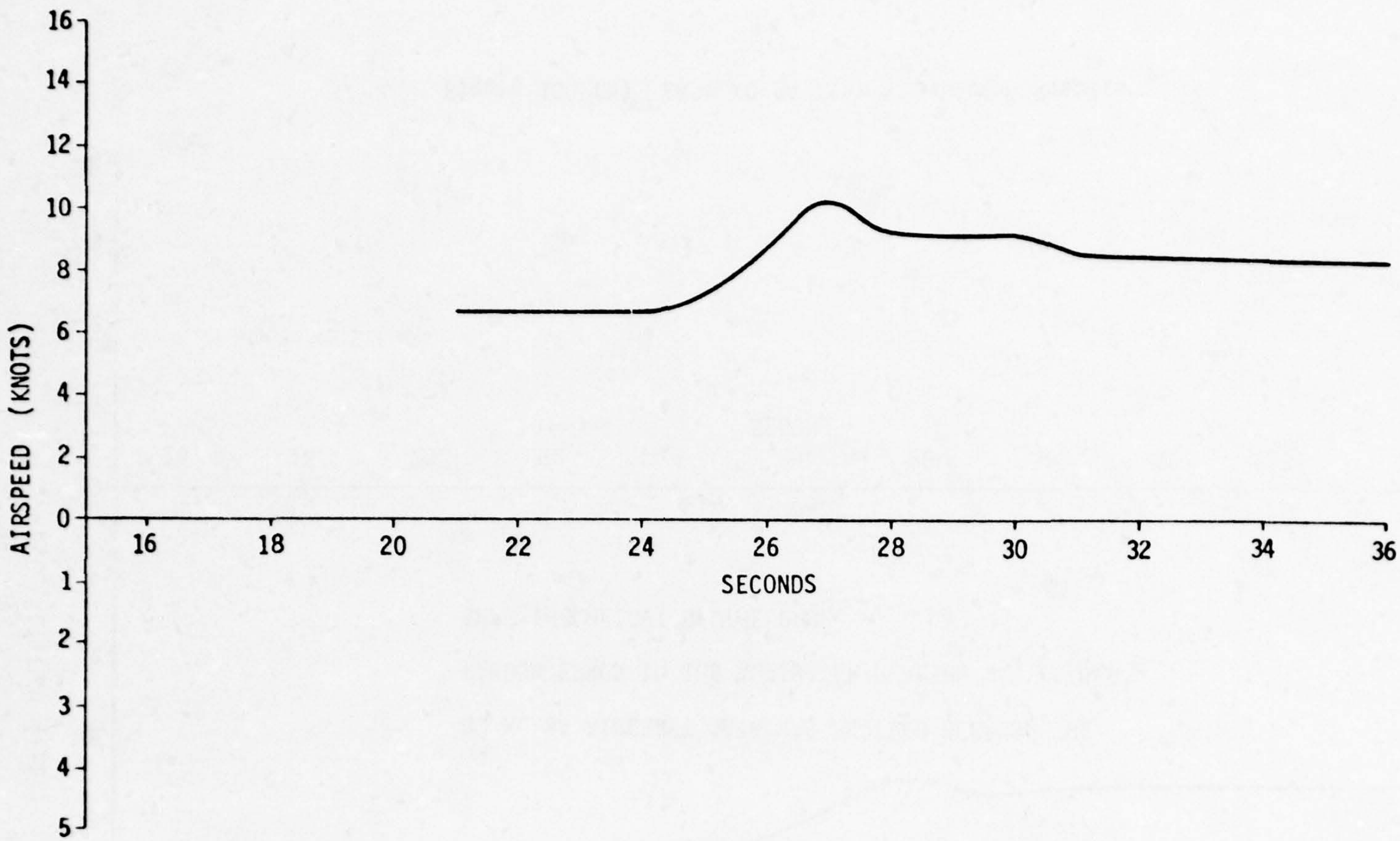


Figure 160 (c) EASY XC-8A Taxi Simulation Results

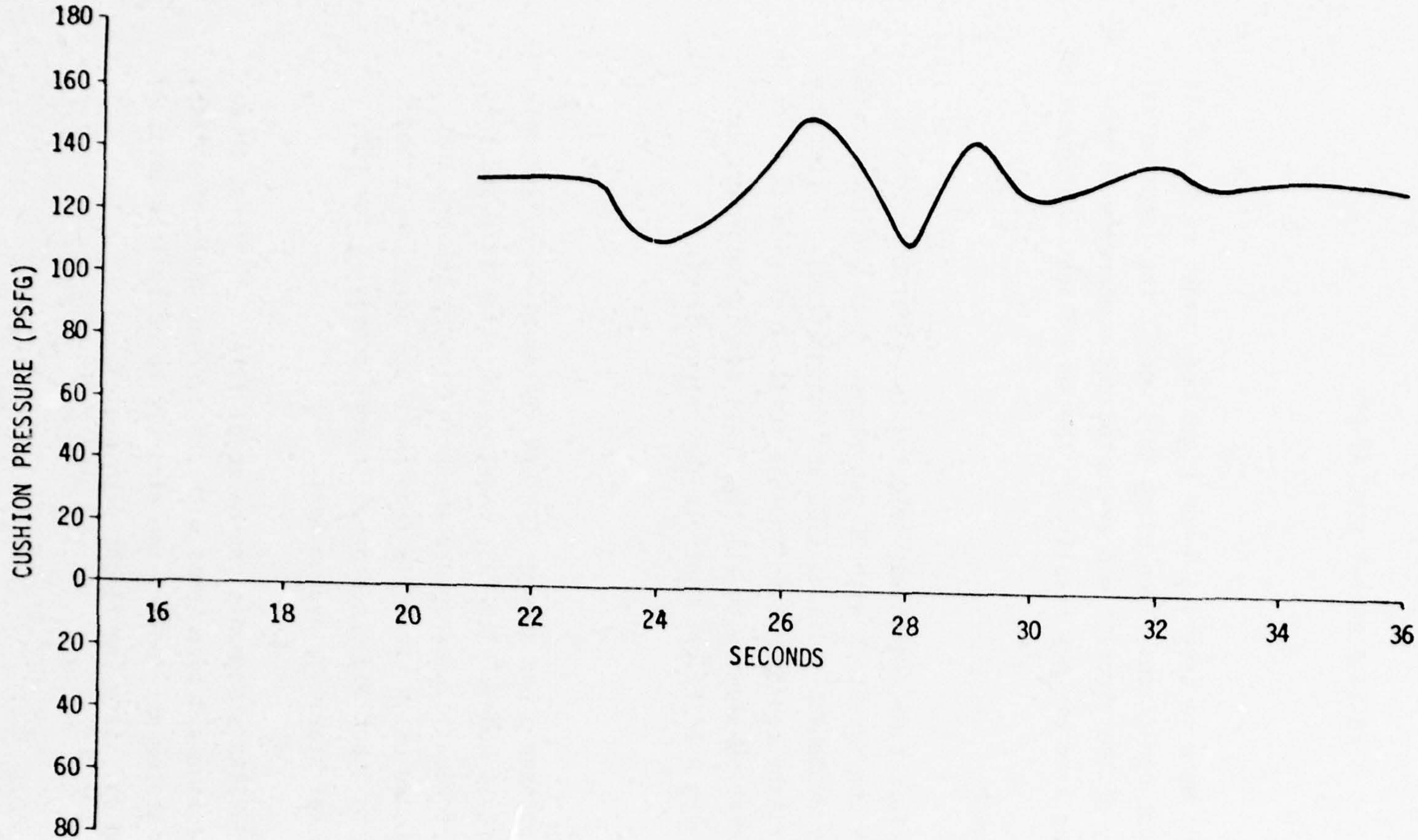


Figure 160 (d) EASY XC-8A Taxi Simulation Results

SECTION XII

YC-14 AIRCRAFT SIMULATION

12.1 Objectives

This task's objectives were to create a 6-DOF rigid body model of the YC-14 aircraft for one flight condition. Then using this model, the longitudinal and lateral stability of the aircraft was evaluated and compared with the YC-14 flight simulator time history results for the same flight configuration.

12.2 Technical Approach

The YC-14 flight configuration, which was selected for simulation, was a conventional approach configuration with 30 deg flaps, at an altitude of 500 ft. with no stability augmentation (i.e. CTOL and unaugmented). A complete listing of the trim flight configuration for the YC-14 simulation is given in Table 15. This information along with all the YC-14 aerodynamic data was obtained from the Boeing C-14 Flight Controls Technology Staff.

12.2.1 Model File

Five standard EASY components and several FORTRAN statements were assembled to form the EASY 6-DOF YC-14 model. The EASY model description file and the resulting EASY model schematic diagram are shown in Figures 161 and 162. Components VA, OL, DL and DS in this file form the 6 DOF aerodynamic model. Component TB is used to input the elevator and rudder schedules for the longitudinal and lateral stability evaluations.

There are also five FORTRAN statements in the model file. The first three statements update the $\sin\alpha$ and $\cos\alpha$ terms with the current angle of attack. The last two Fortran statements convert the aircraft velocity from units of ft/sec into knots and calculate the flight path angle γ.

TABLE 15 YC-14

TRIM FLIGHT CONFIGURATION

(CTOL AND UNAUGMENTED)

FLAPS	30 DEG.
TRUE AIRSPEED	120 KNOTS
ALTITUDE	500 FT
GROSS WEIGHT	160000 LBS
CENTER OF GRAVITY (c.g.)	32% MAC
MOMENTS OF INERTIA:	
IXX	1.8×10^6 SLUGS-FT^2
IYY	3.3×10^6 SLUGS-FT^2
IZZ	4.2×10^6 SLUGS-FT^2
IXZ	0.5×10^6 SLUGS-FT^2
THRUST	26,700 LBS. TOTAL
ATTITUDE:	
ANGLE OF ATTACK (α)	8.63 DEG.
SIDE SLIP ANGLE (β)	0.0 DEG.
ANGLE OF CLIMB (γ)	-3.00 DEG.
WING AREA (s)	1762 FT^2
WING SPAN (b)	129 FT
MEAN CHORD LENGTH (c)	14.68 FT
LONGITUDINAL CONTROL SURFACES:	
ELEVATOR	0.0 DEG.
STABILIZER	-2.39 DEG.
SPOILER	0.0 DEG.
LATERAL CONTROL SURFACES:	
RUDDER	0.0 DEG.
AILERION	0.0 DEG.

```
MODEL DESCRIPTION        YC-14 FLIGHT SIMULATION (CTOL AND UNAUGMENTED)
ADD PARAMETERS=VKNOTS,GAMMA,ALTRATE
LOCATION=57       VA      INPUTS=DS
FORTRAN STATEMENTS
C****************************************
C****************************************
C  THESE FORTRAN STATEMENTS UPDATE THE COS(ALPHA) AND SIN(ALPHA)
C  WITHOUT CHANGING THE VALUE OF ALSVA.
C  CAUTION  THESE FORTRAN STATEMENTS CANNOT BE USED WHEN THE
C         AERODYNAMIC DERIVATIVES ARE REFERENCED TO THE BODY AXIS.
      RPD=.01745329
      CALVA=COS(AL VA*RPD)
      SALVA=SIN(AL VA*RPD)
C****************************************
C****************************************
      VKNOTS=VT VA/1.6889
      GAMMA=PITDS-AL VA
LOCATION=14       TB
LOCATION=34       OL      INPUTS=VA,TB(A2=ELE)
LOCATION= 1       DL      INPUTS=VA,OL,TB(B2=RUD)
LOCATION=40       DS      INPUTS=OL,DL
FORTRAN STATEMENTS
      ALTRATE=XDOT(10)*60.
END OF MODEL
PRINT
```

Figure 161 YC-14 Model File

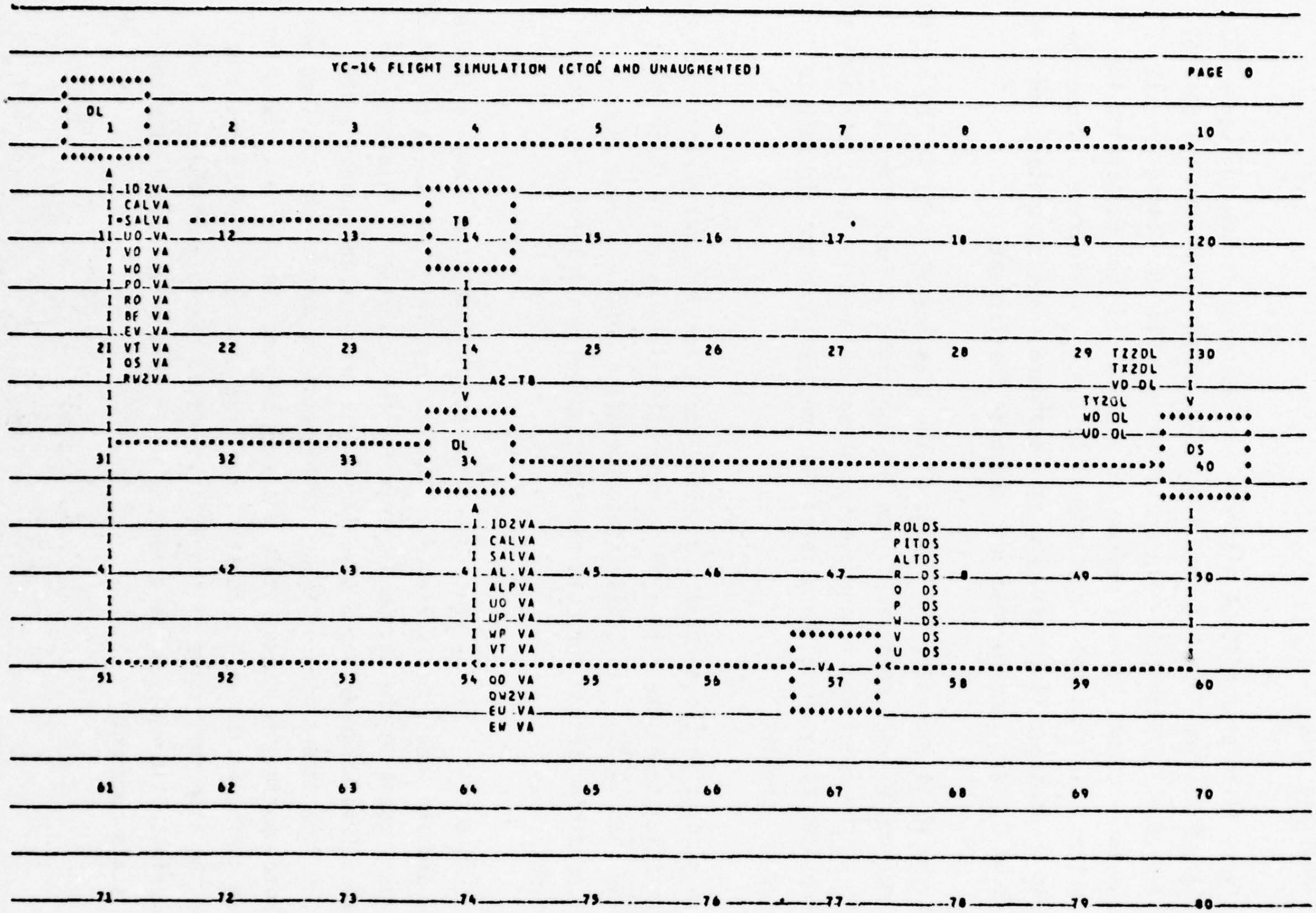


Figure 162 YC-14 Flight Simulation EASY Model Schematic Diagram

12.2.2 Analysis File

The analysis file for the YC-14 aircraft simulation is listed in Figure 163. A tabularized listing of the longitudinal and lateral stability and control derivatives specified in this file are given in Table 16. These are all non-dimensional stability axis referenced values. The box containing the single asterisk in this table corresponds to the C_{xs} derivative. Since the component OL does not list this term as a standard input parameter, the drag produced by C_{xs} was incorporated into the bias coefficient.

To trim the YC-14 aircraft, the values of the bias coefficients XO, ZO and MO in the math model were calculated from the equations of motion for the aircraft at the specified flight configuration. A detailed description of the procedure used to evaluate the bias coefficients to trim the aircraft is presented in Appendix A.

Prior to simulating the aircraft, a steady state analysis was performed on the model to "fine tune" the values of each of active states until their rates were less than 10^{-4}. This steady state operating point was used as the initial condition vector for both the longitudinal and lateral stability evaluations.

In order to obtain a steady state operating point it was necessary to freeze the yaw and altitude state variables. The yaw was frozen because a column of zeros was present in the Jacobean matrix for the yaw state. This made the Jacobean matrix singular and a singular matrix prevents the steady state algorithm from determining a steady state operating point. The altitude was also frozen during the steady state analysis because the aircraft was in an approach configuration with a constant non-zero sink rate. Since the sink rate is the time derivative of the altitude (i.e. the rate of the altitude state variable) it would be incorrect to drive this rate to zero with the steady state command.

```
TABLE,A2TTB,6                              } ELEVATOR
0.0,0.5,0.6,3.4,3.5,20.0                   } SCHEDULE
0.0,0.0,-1.25,-1.25,0.0,0.0                }
TABLE,B2TTH,2                              } RUDDER
0.0,20.0                                   } SCHEDULE
0.0,0.0                                    }
PARAMETER VALUES
AILDL=0.0
IDIVA=3,VS VA=202.67,ALSVA=8.53,S  VA=1762.0,UW VA=0.0,VW VA=0.0
WW VA=0.0,PW VA=0.0,QW1VA=0.0,RW1VA=0.0,IDGVA=6
XO DL=-.10612,XA DL=-1.01
XU DL=-.485,XDEDL=-.237,XTRDL=0.0,XGEDL=0.0
KXBDL=1.0,ZO DL=-1.93459,ZA DL=-7.04,ZADDL=-4.5,ZQ DL=-8.4
ZU DL=.397,ZDEDL=-1.536,ZTRDL=0.0,ZGEDL=0.0
KZBDL=1.0,ZDSDL=-1.204,MO DL=-.23086,MALDL=-.918
MADDL=-35.0,MQ DL=-57.4,MU DL=.152,MDEDL=-7.093
MTRDL=0.0,MGEDL=0.0,KMBDL=1.0,MDSDL=-5.535
MB DL=0.0,KGEDL=0.0,MA1DL=4972.96,C  DL=14.68,XP1DL=0.0
ISWDL=3,FX1DL=0.0,FZ1DL=0.0,TY1DL=0.0,STADL=-2.39
YB DL=-1.455,YBDDL=.191,YP DL=.0902,YR DL=.60,YDRDL=.658
YDADL=0.0,KCYDL=1.0,YTRDL=0.0,YFSDL=0.0,YGEDL=1.0,KYBDL=1.0
YBRDL=1.0,LB DL=-.109,LBDDL=.0136,LP DL=-.634,LR DL=.428
LDRDL=.059,LDADL=.0974,KCLDL=1.0,LTRDL=0.0,LFSDL=0.0
LGEDL=1.0,KLBDL=1.0,LBRDL=1.0,NB DL=.273,NBDDL=-.0915
NP DL=-.140,NR DL=-.379,NDRDL=-.327,NDADL=0.0
NTRDL=0.0,NFSDL=0.0,NGEDL=1.0,KNBDL=1.0,NBRDL=1.0,FSPDL=0.0
FY1DL=0.0,TX1DL=0.0,TZ1DL=0.0,B  DL=129.0
IXXDS=1800000.0,IYYDS=3300000.0,IZZDS=4200000.0,IXZDS=500000.0
INITIAL CONDITIONS=U  DS=200.38,W  DS=30.41
PITDS=5.63,ALTDS=500.0
ERROR CONTROLS=U  DS=.001,V  DS=.001,W  DS=.001,P  DS=.001
Q  DS=.001,R  DS=.001,ROLDS=.001,PITDS=.001,YAWDS=.001,ALTDS=.001
PRINT CONTROL=7
ALL STATES
LINEAR ANALYSIS
INT CONTROLS=YAWDS=0,ALTDS=0
LINEAR ANALYSIS
STEADY STATE
XIC-X,XIC1-XIC
LINEAR ANALYSIS
PRINT CONTROL=4
PRINTER PLOTS,PLOT ON
PLOT ID=     PAUL R. PERKINS  MS 47-03
```

Figure 163 YC-14 Analysis File

```
DISPLAY1
ALTDS,VS,TIME
ALTRATE,VS,TIME
VKNOTS,VS,TIME
A2 TB,VS,TIME          LONGITUDINAL
DISPLAY2               PRINTER
GAMMA,VS,TIME          PLOTS
PITDS,VS,TIME
AL VA,VS,TIME
Q  DS,VS,TIME
TITLE     YC-14 LONGITUDINAL STABILITY EVALUATION
NO STATES
INT CONTROLS
W  DS=1,U  DS=1,Q  DS=1,PITDS=1
ALTDS=1
TINC=.1,TMAX=30.0,INT MODE=2,PRATE=2,PRINT CONTROL=4
LINEAR ANALYSIS
SIMULATE
XIC-X
LINEAR ANALYSIS
NO STATES
XIC-XIC1
INT CONTROLS
V  DS=1,ROLDS=1,YAWDS=1,P  DS=1,R  DS=1
ALTDS=1
TABLE,A2TTB,2      ELEVATOR
0.0,20.0           SCHEDULE
0.0,0.0
TABLE,B2TTB,6
0.0,0.5,0.6,2.6,2.7,20.0     RUDDER
0.0,0.0,7.5,7.5,0.0,0.0      SCHEDULE
DISPLAY1
ROLDS,VS,TIME
P  DS,VS,TIME
BE VA,VS,TIME
VKNOTS,VS,TIME      LATERAL
DISPLAY2            PRINTER
YAWDS,VS,TIME       PLOTS
R  DS,VS,TIME
B2 TB,VS,TIME
ALTDS,VS,TIME
TITLE     YC-14 LATERAL STABILITY EVALUATION
LINEAR ANALYSIS
SIMULATE
XIC-X
LINEAR ANALYSIS
```

Figure 163   YC-14 Analysis File (Cont.)

TABLE 16

YC-14 Longitudinal Stability and Control Derivatives: OL

	BIAS	ALPHA	U	$\dot{\text{ALPHA}}$	Q	δ ELEV	TRUNK	SPOILERS	GROUND EFFECT	LARGE SIDE SLIP	STABIL-IZER	BETA
FX (DRAG)	-.10612 XO	-1.01 XA	-.485 XU			-.237 XDE	0.0 XTR	0.0 XSP	0.0 XGE	1.0 KXB	*	
FZ (LIFT)	-1.9346 ZO	-7.04 ZA	+.397 ZU	-4.50 ZAD	-8.40 ZQ	-1.536 ZDE	0.0 ZTR	0.0 ZSP	0.0 ZGE	1.0 KZB	-1.204 ZDS	
TY=M (PITCH)	-.23086 MO	-.918 MAL	+.152 MU	-35.0 MAD	-57.4 MQ	-7.093 MDE	0.0 MTR	0.0 MSP	0.0 MGE	1.0 KMB	-5.535 MDS	0.0 MB

$^{*}C_{XS} = -.178$

Lateral Stability and Control Derivatives: OL

	BETA	$\dot{\text{BETA}}$	P	R	δ RUD	δ AIL	TRUNK	SPOILERS	GROUND EFFECT	LARGE SIDE SLIP	β_{RUD}	AERO ELASTIC-ITY
FY (SIDE)	-1.455 YB	+.191 YBD	+.0902 YP	+.60 YR	+.658 YDR	0.0 YDA	0.0 YTR	0.0 YFS	1.0 YGE	1.0 KYB	1.0 YBR	1.0 KCY
TX=L (ROLL)	-.109 LB	+.0136 LBD	-.634 LP	+.428 LR	+.059 LDR	+.0974 LDA	0.0 LTR	0.0 LFS	1.0 LGE	1.0 KLB	1.0 LBR	1.0 KCL
TZ=N (YAW)	+.273 NB	-.0915 NBD	-.140 NP	-.379 NR	-.327 NDR	0.0 NDA	0.0 NTR	0.0 NFS	1.0 NGE	1.0 KNB	1.0 NBR	

(THESE DERIVATIVES ARE ALL STABILITY AXIS, NONDIMENSIONAL VALUES)

12.3 Longitudinal Stability Evaluation

The longitudinal stability of the YC-14 was evaluated by perturbing the aircraft from its trim flight path with an elevator "kick". The model was initialized for this analysis by turning off all the lateral state variables and by setting the initial condition vector to the steady state operating point. A -1.25 degree, 3 second elevator impulse like deflection was input by Table A2TTB. This was the same elevator impulse which was evaluated by the YC-14 flight simulator.

The initial condition linear analysis for the longitudinal stability evaluation is given in Figure 164. The stability matrix generated during the linear analysis correlates very well with the stability matrix output by the YC-14 flight simulator.

Since the initial conditions and the stability matrix were in good agreement with the YC-14 flight simulator, the EASY model was ready to simulate the YC-14's response to a -1.25 degree, 3 second elevator impulse. The resulting EASY YC-14 model time histories compared with the YC-14 Flight Simulator time histories for altitude, airspeed, pitch and pitch rate are shown respectively in Figures 165-a, -b, -c, and -d. The correlation of the EASY simulation results and the YC-14 flight simulator results for the first 10 seconds is considered good. After 10 seconds the two simulations tend to diverge. This slight divergent trend can be attributed to the limitations of the EASY constant coefficient model.

12.4 Lateral Stability Evaluation

The lateral stability of the YC-14 was evaluated by perturbing the aircraft from its trim path with a rudder "kick". To initialize the model for this analysis, all the longitudinal states were frozen and the initial condition vector reset to the same steady state operating point used in the longitudinal

/*/*/*/ LINEAR ANALYSIS /*/*/*/

STATE NAME	OPERATING POINT	PERTURBATION SIZE	INTEGRATOR CONTROL
1 U DS	201.15	.100E-02	1
2 V DS	0.	.100E-02	0
3 W DS	30.155	.100E-02	1
4 P DS	0.	.100E-02	0
5 Q DS	.27737E-17	.100E-02	1
6 R DS	0.	.100E-02	0
7 ROLDS	0.	.100E-02	0
8 PITDS	5.2813	.100E-02	1
9 YAWDS	0.	.100E-02	0
10 ALTDS	500.00	.100E-02	1

RATES AT OPERATING POINT

1 U DS = .45169E-06	2 V DS = 0.	3 W DS = -.19345E-05	4 P DS = 0.	5 Q DS = -.60197E-06
6 R DS = 0.	7 ROLDS = 0.	8 PITDS = .27737E-17	9 YAWDS = 0.	10 ALTDS = -11.512

8 ELEMENTS OF /RATIO/ DIFFER FROM 1 BY 10%. THESE ELEMENTS ARE PRECEDED BY AN * IN THE STABILITY MATRIX

RATIO(1, 1) = 1.27362
RATIO(1, 2) = .892038
RATIO(1, 5) = 1.99460
RATIO(2, 4) = .791827
RATIO(2, 5) = 1.90823
RATIO(3, 2) = .493252
RATIO(3, 4) =-12.7721
RATIO(3, 5) = 2.03240

STABILITY MATRIX

	U DS	W DS	Q DS	PITDS	ALTDS
U DS	*-.5595E-01	* .1405	-.5211	-.5744	*-.1539E-01
W DS	-.1862	-.6049	3.383	*-.4314E-01	* .9886E-02
Q DS	.6118E-01	*-.7815E-02	-1.255	*-.2876E-03	* .3836E-02
PITDS	0.	0.	1.000	0.	0.
ALTDS	.9205E-01	-.9958	0.	3.544	0.

5 EIGENVALUES

	REAL	IMAGINARY	NATURAL FREQ.	DAMPING RATIO
1	.409570E-02	+- .149233	.149289	-.274347E-01
2	-.114645	0.	.114645	1.00000
3	-.634103	0.	.634103	1.00000
4	-1.17482	0.	1.17482	1.00000

Figure 164 YC-14 Longitudinal Stability Evaluation Linear Analysis

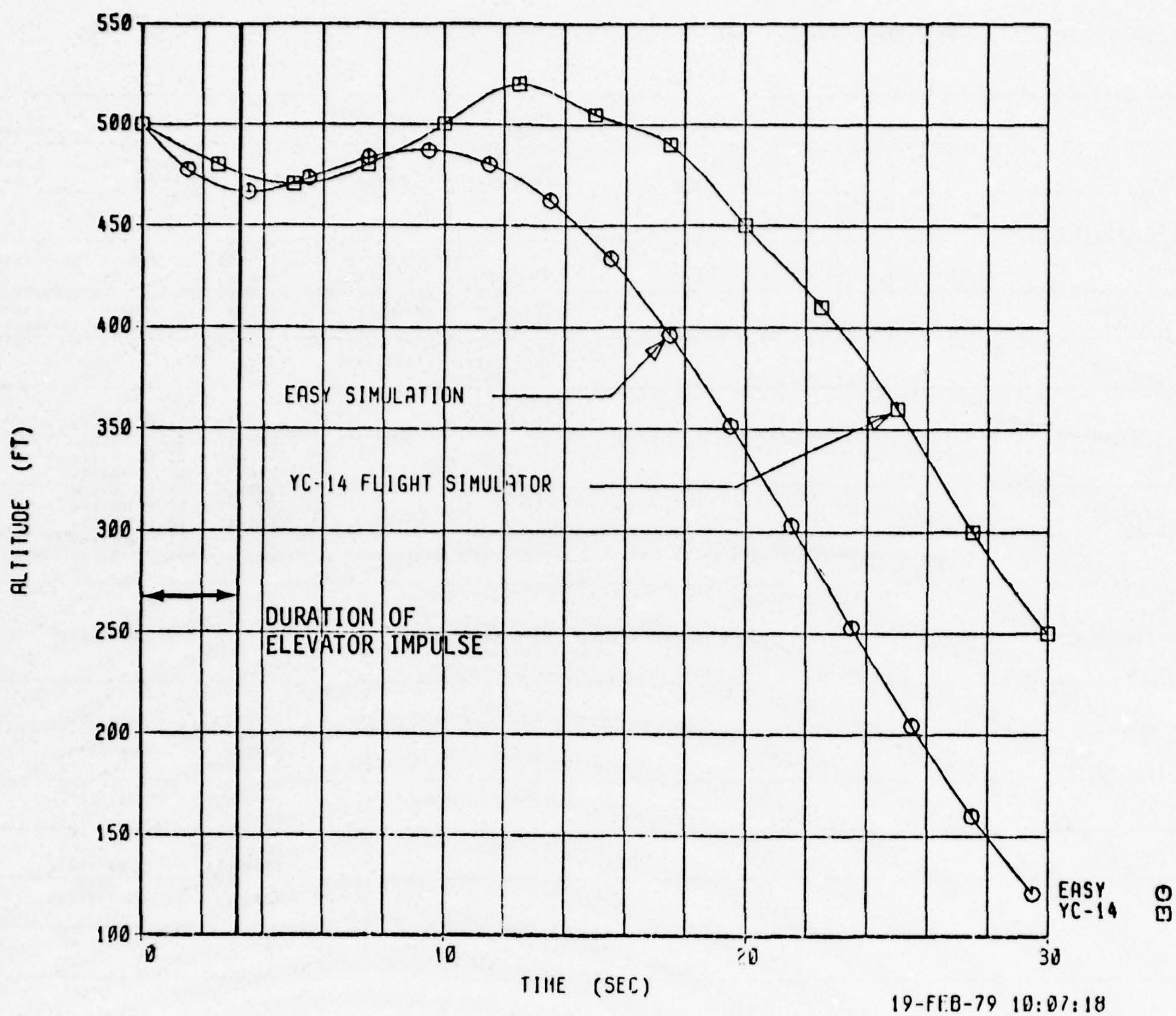


Figure 165-(a) YC-14 Longitudinal Stability Evaluation
A Comparison of the EASY Time History Simulation
Results with the YC-14 Flight Simulator Results

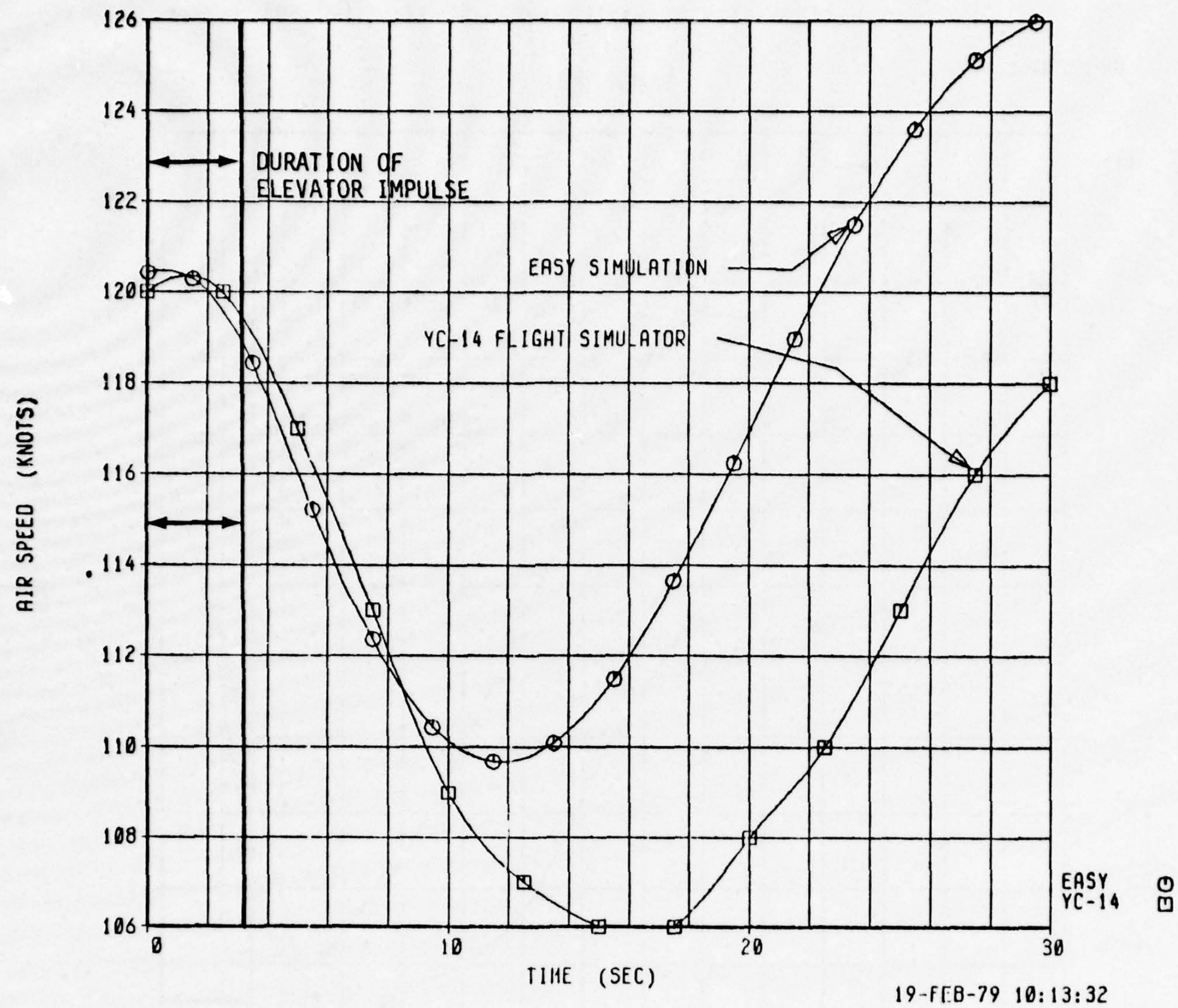


Figure 165-(b) YC-14 Longitudinal Stability Evaluation
A Comparison of the EASY Time History Simulation
Results with the YC-14 Flight Simulator Results

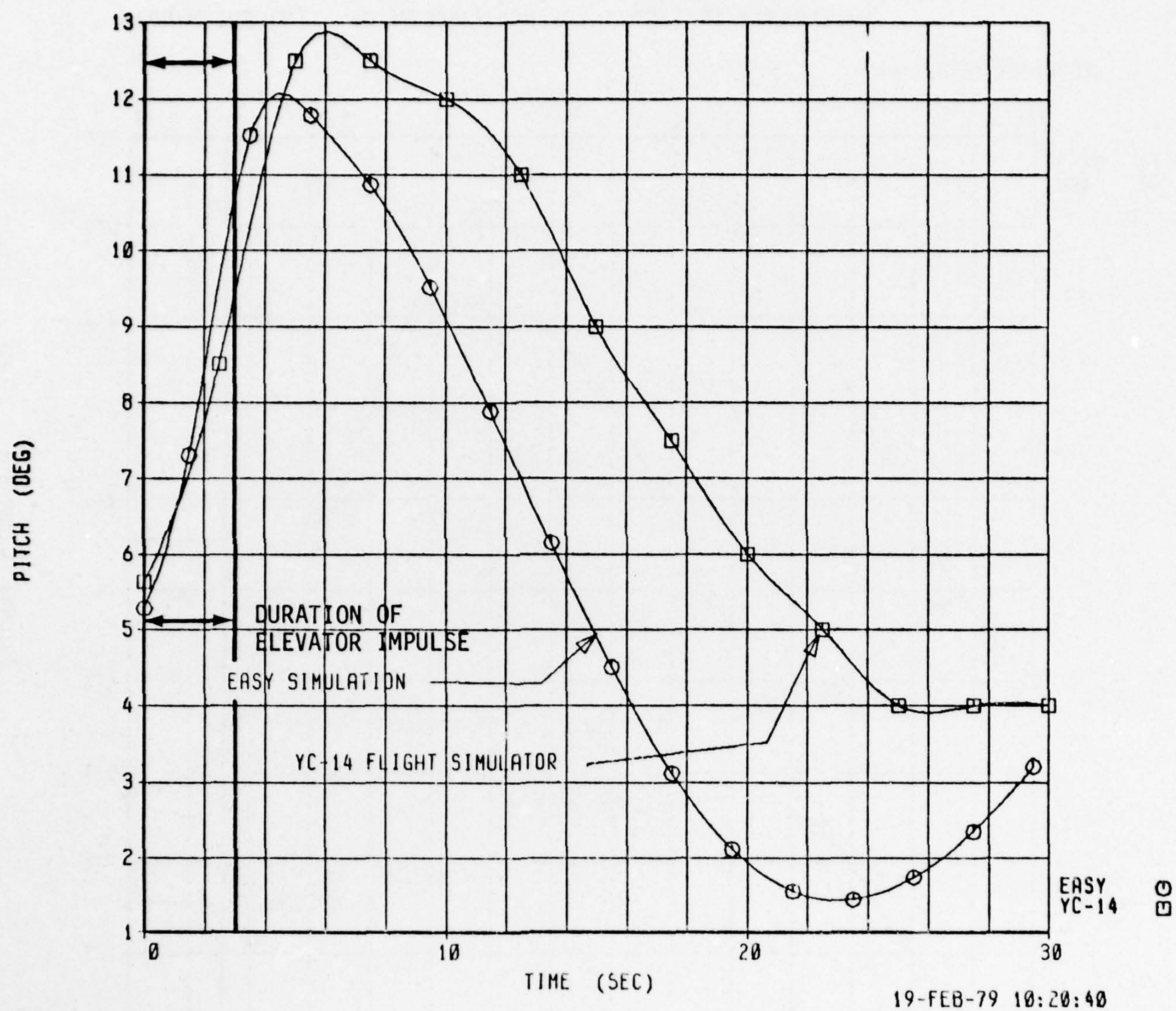


Figure 165-(c) YC-14 Longitudinal Stability Evaluation
A Comparison of the EASY Time History Simulation
Results with the YC-14 Flight Simulator Results

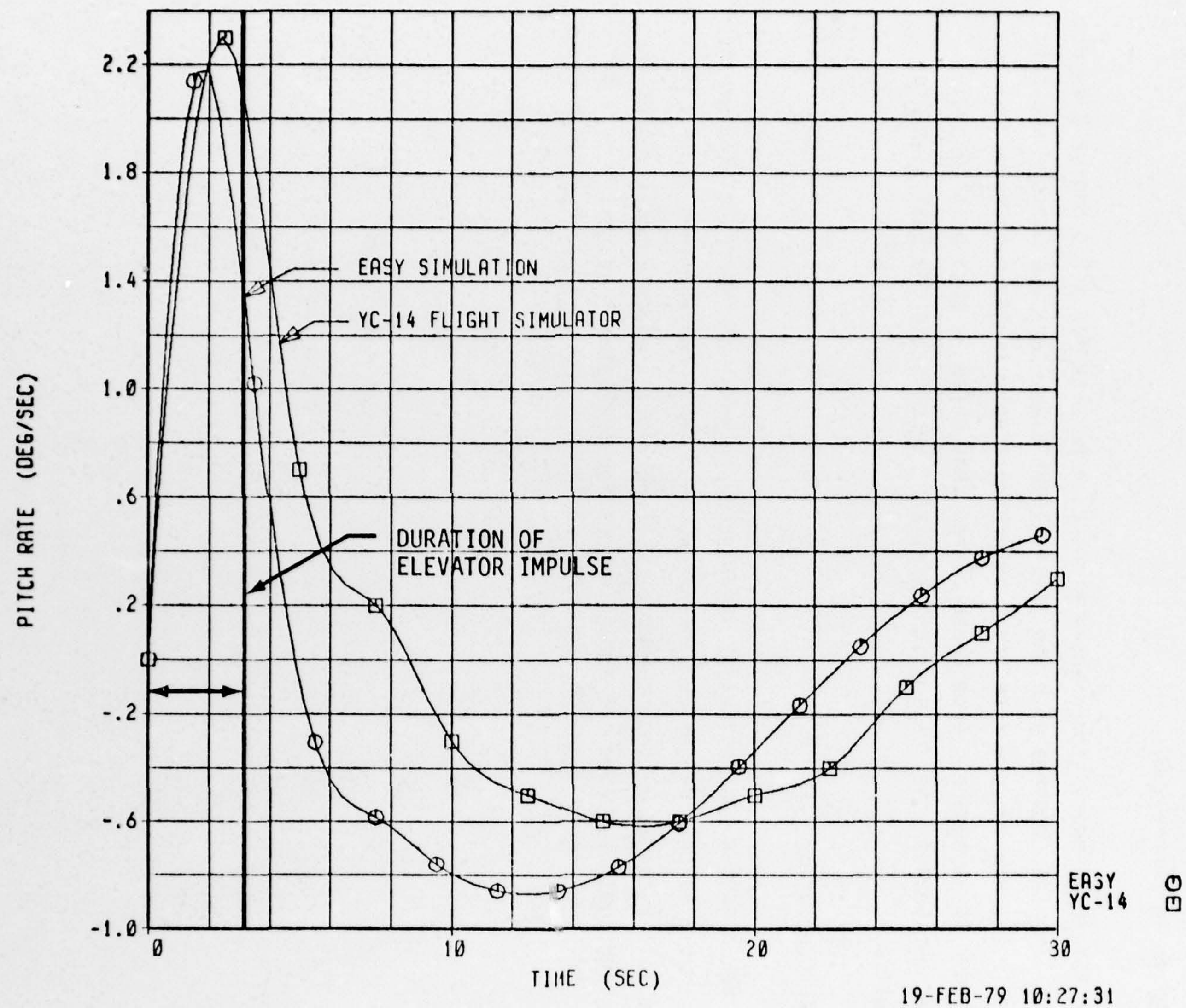


Figure 165-(d) YC-14 Longitudinal Stability Evaluation
A Comparison of the EASY Time History Simulation
Results with the YC-14 Flight Simulator Results

analysis. A 7.5 degree, 2 second rudder impulse was input by Table B2TTB. The linear analysis of the lateral model at the steady state initial condition is given in Figure 166.

The EASY YC-14 model time history simulation results for a 7.5 degree 2 second rudder impulse compared with the YC-14 Flight Simulator time history plots for beta, roll, yaw and yaw rate are shown respectively in Figures 167-a, -b, -c and -d. It is apparent from these curves that there is good point by point agreement between the YC-14 flight simulator results and the EASY simulator results.

12.5 Conclusion

The YC-14 EASY 6-DOF EASY Model is an accurate model of the YC-14 aircraft at the specified trim flight configuration. For small pertubations about this trim flight configuration, the model will reliably predict general trends for the longitudinal states and actual values for the lateral states.

/*/*/*/ LINEAR ANALYSIS /*/*/*/

STATE NAME	OPERATING POINT	PERTURBATION SIZE	INTEGRATOR CONTROL
1 U DS	201.15	.100E-02	0
2 V DS	0.	.100E-02	1
3 W DS	30.155	.100E-02	0
4 P DS	0.	.100E-02	1
5 Q DS	.27737E-17	.100E-02	0
6 R DS	0.	.100E-02	1
7 ROLDS	0.	.100E-02	1
8 PITDS	5.2813	.100E-02	0
9 YAWDS	0.	.100E-02	1
10 ALTDS	500.00	.100E-02	1

RATES AT OPERATING POINT

1 U DS	= .45169E-06	2 V DS	= 0.	3 W DS	= -.19345E-05	4 P DS	= 0.	5 Q DS	= -.60197E-06
6 R DS	= 0.	7 ROLDS	= 0.	8 PITDS	= .27737E-17	9 YAWDS	= 0.	10 ALTDS	= -11.512

1 ELEMENTS OF /RATIO/ DIFFER FROM 1 BY 10/. THESE ELEMENTS ARE PRECEDED BY AN * IN THE STABILITY MATRIX
RATIO(6, 4) = .500000

STABILITY MATRIX

	V DS	P DS	R DS	ROLDS	YAWDS	ALTDS
V DS	-.1235	.5290	-3.471	.5620	0.	0.
P DS	-.2116	-1.385	.6572	.5238E-02	0.	0.
R DS	.1649	-.3287	-.1346	-.1103E-01	0.	0.
ROLDS	0.	1.000	.9244E-01	.4475E-20	0.	0.
YAWDS	0.	0.	1.004	.4862E-19	0.	0.
ALTDS	0.	0.	0.	* .4573E-05	0.	0.

6 EIGENVALUES

	REAL	IMAGINARY	NATURAL FREQ.	DAMPING RATIO
1	.444518E-01	0.	.444518E-01	-1.00000
2	0.	0.	0.	0.
3	0.	0.	0.	0.
4	-.164711	+- .933129	.947554	.173827
5	-1.35823	0.	1.35823	1.00000

Figure 166 YC-14 Lateral Stability Evaluation Linear Analysis

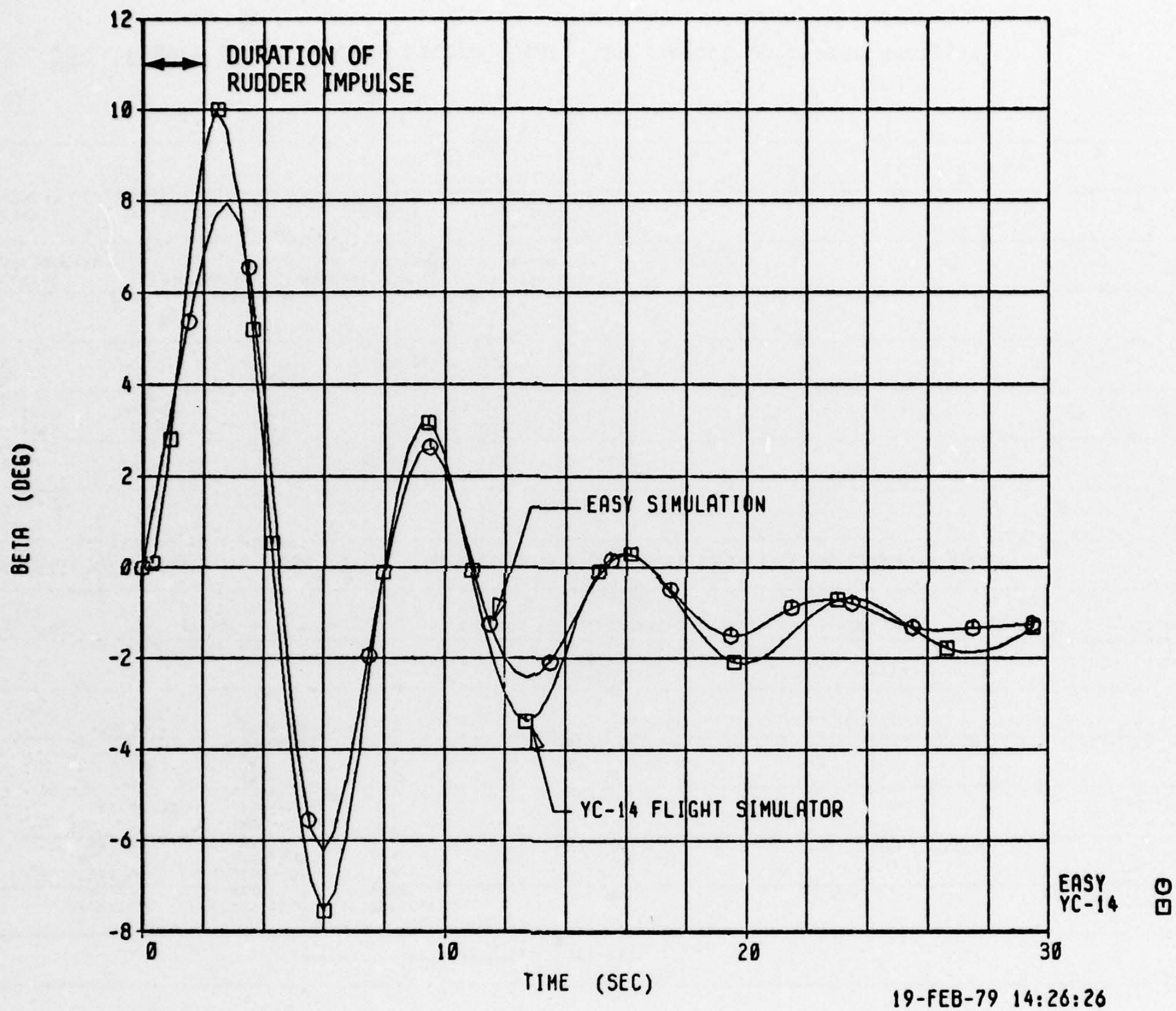


Figure 167-(a) YC-14 Lateral Stability Evaluation
A Comparison of the EASY Time History Simulation
Results with the YC-14 Flight Simulator Results

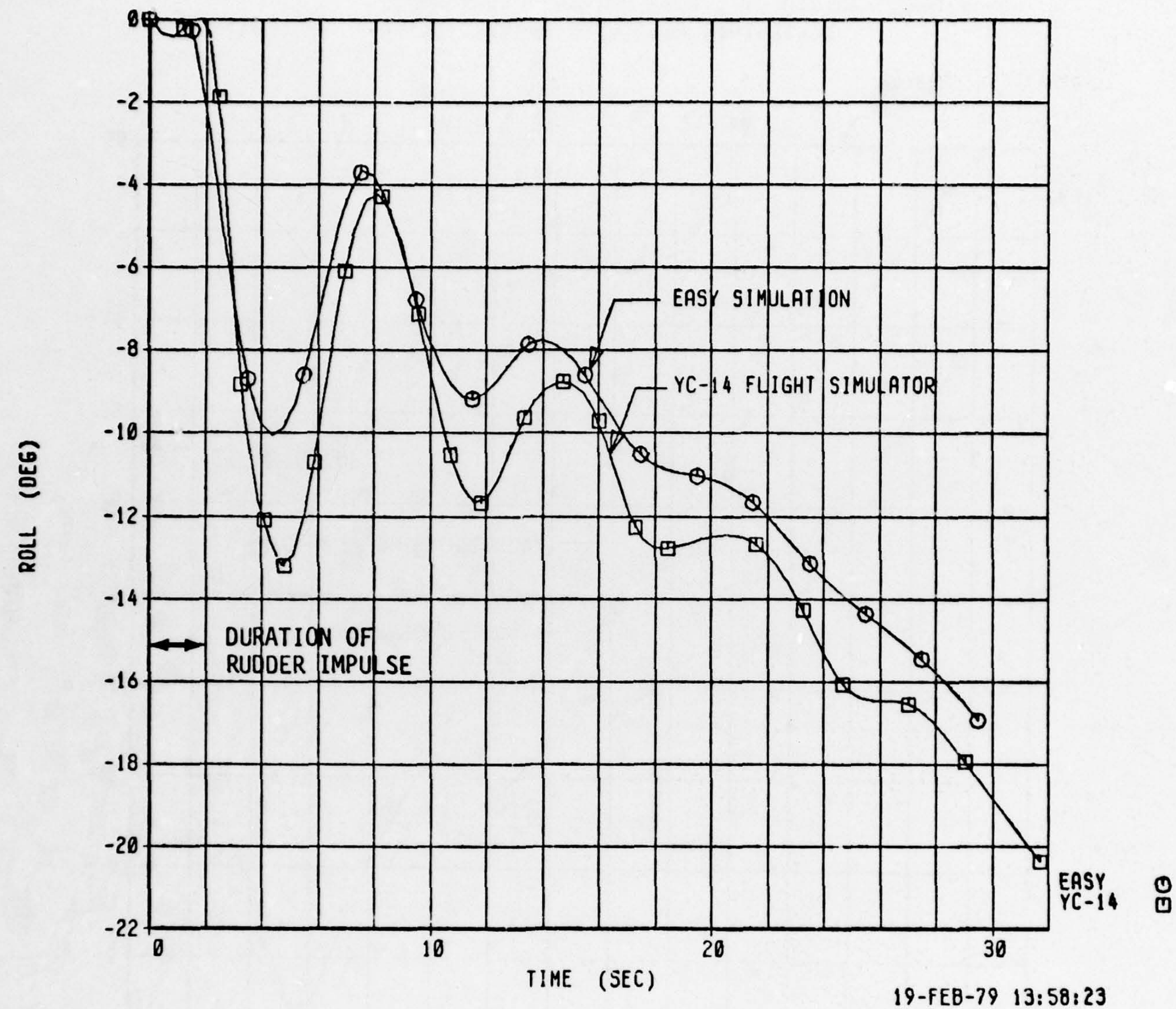


Figure 167-(b) YC-14 Lateral Stability Evaluation
A Comparison of the EASY Time History Simulation
Results with the YC-14 Flight Simulator Results

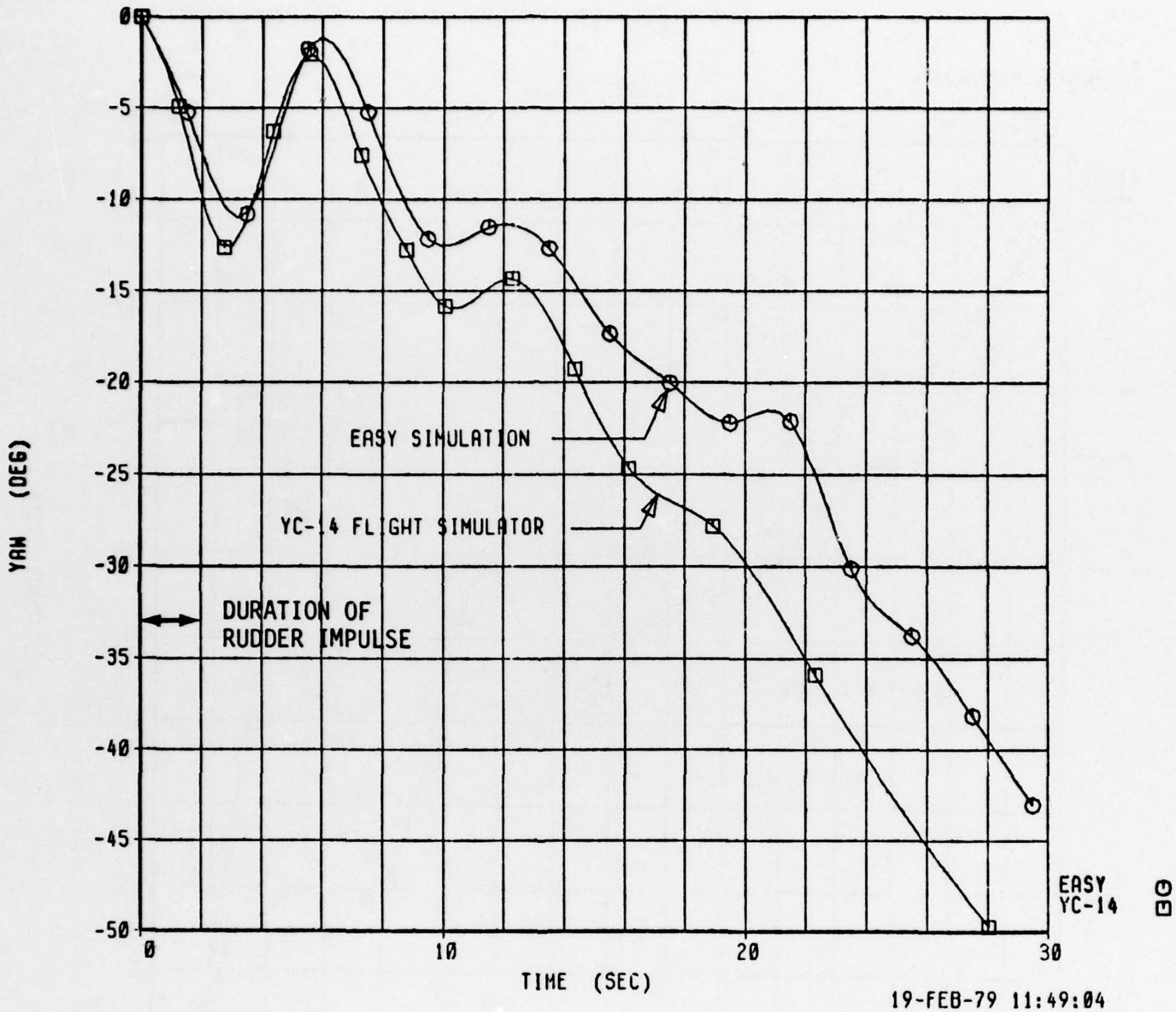


Figure 167-(c) YC-14 Lateral Stability Evaluation
A Comparison of the EASY Time History Simulation
Results with the YC-14 Flight Simulator Results

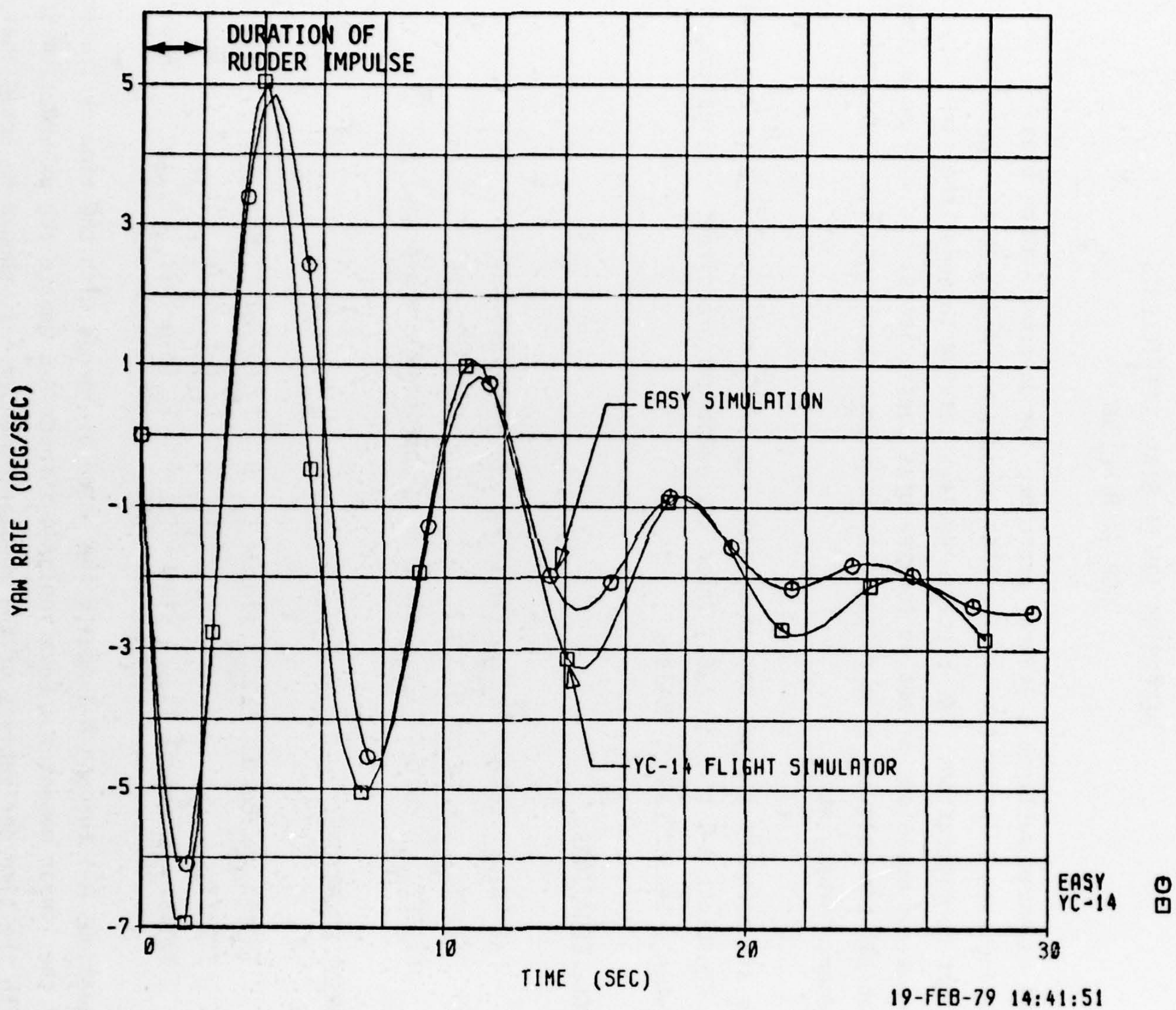


Figure 167-(d) YC-14 Lateral Stability Evaluation
A Comparison of the EASY Time History Simulation
Results with the YC-14 Flight Simulator Results

APPENDIX A

DETERMINATION OF BIAS COEFFICIENTS FOR TRIM FLIGHT

This appendix outlines a general procedure for determining the bias coefficients in order to trim the EASY aerodynamic model at the specified flight configuration. The process involves solving three of the six parametric equations of motion for the bias coefficients. It will be assumed that all the relevant trim flight data have been specified (i.e., elevator, rudder, thrust, etc.).

The six parametric equations of motion, resolved into the body axis are given in Equations A-1 through A-6. These six equations are direct application of Newton's Second Law of Motion.

FORCE EQUATIONS:

$$FX1+FX_{AERO}\cos\alpha - FZ_{AERO}\sin\alpha - mg\sin\theta = m(\dot{u}+qw-rv) \qquad \text{A-1}$$

$$FY1+FY_{AERO} + mg\cos\theta\sin\phi = m(\dot{v}+ru-pw) \qquad \text{A-2}$$

$$FZ1+FZ_{AERO}\cos\alpha + FX_{AERO}\sin\alpha + mg\cos\theta\cos\phi = m(\dot{w}+pv-qu) \qquad \text{A-3}$$

MOMENT EQUATIONS:

$$TX1+TX_{AERO}\cos\alpha - TZ_{AERO}\sin\alpha = I_{XX}\dot{p} - I_{XZ}(\dot{r}+pq)-(I_{YY}-I_{ZZ})qr \qquad \text{A-4}$$

$$TY1+TY_{AERO} = I_{YY}\dot{q} - I_{XZ}(r^2-p^2)-(I_{ZZ}-I_{XX})rp \qquad \text{A-5}$$

$$TZ1+TZ_{AERO}\cos\alpha + TX_{AERO}\sin\alpha = I_{ZZ}\dot{r} - I_{XZ}(\dot{p}-qr)-(I_{XX}-I_{YY})pq \qquad \text{A-6}$$

Equations A-1 through A-3 equate the external forces with the time derivatives of the linear momentum. Equations A-4 through A-6 equate the external torques with the time derivatives of the angular momentum. It should be noted that the terms FX1, FY1, FZ1, TX1, TY1, and TZ1 in these equations represent the total of all external forces and moments except for aerodynamic terms (i.e., FX_{AERO}, FY_{AERO}, etc.) and gravitational terms ($mg\sin\theta$, $mg\cos\theta\sin\phi$

etc.) which are implicit in these equations. Typically these other external forces and moments will be those produced by the engine, landing gear, drag chute, etc.

Equations A-1, A-3 and A-5 are needed to solve for the bias coefficients XO, ZO and MO respectively. These bias coefficients are constant terms in FX_{AERO}, FZ_{AERO} and TY_{AERO}. The procedure to evaluate these coefficients is outlined below. For a definition of the symbols used in the following equations refer to components VA, OL, DL and SG in the Users Manual.

1. Select the appropriate set of dimensional or non-dimensional equations for FX_{AERO}, FY_{AERO} and TY_{AERO} listed here and substitute them into Equation A-1, A-3 and A-5 respectively.

   DIMENSIONAL EQUATIONS:

$$FX_{AERO} = (XO + XA \cdot WP + XU \cdot UP + XDE \cdot ELE + XTR + XSP \cdot SPO + XGE \cdot KGE) \cdot KXB$$

$$FZ_{AERO} = (ZO + ZA \cdot WP + ZAD \cdot (\dot{W} + \dot{W}W) + ZQ \cdot QO + ZU \cdot UP + ZDE \cdot ELE + ZTR + ZSP \cdot SPO + ZGE \cdot KGE + ZDS \cdot STA) \cdot KZB$$

   where

$$\dot{W} = WD - UD \cdot SAL$$

$$\dot{W}W = -QW \cdot VT$$

$$TY_{AERO} = (MO + MAL \cdot WP + MAD\ (\dot{W} + \dot{W}W) + MQ \cdot QO + MU \cdot UP + MDE \cdot ELE + MTR + MSP \cdot SPO + MGE \cdot KGE + MDS \cdot STA + MB) \cdot KMB$$

   NON-DIMENSIONAL EQUATIONS:

$$FX_{AERO} = QS \cdot (XO + XA \cdot \widehat{ALP} + XU \cdot UP + XDE \cdot \widehat{ELE} + XTR + XSP \cdot \widehat{SPO} + XGE \cdot KGE) \cdot KXB$$

$$FZ_{AERO} = QS \cdot (ZO + ZA \cdot \widehat{ALP} + (ZAD \cdot (AL\dot{P}HA - \widehat{QW}) + ZQ \cdot \widehat{QO}) \cdot C/(2\ VT) + ZU \cdot UP + ZDE \cdot \widehat{ELE} + ZTR + ZSP \cdot \widehat{SPO} + ZGE \cdot KGE + ZDS \cdot \widehat{STA}) \cdot KZB$$

where

$$\dot{ALPHA} = (WD - \widehat{AL} \cdot UD)/UO$$

$$\widehat{ALP} = ALP\,\pi/180, \text{ etc. for ELE, QW, QO, AL, SPO, STA}$$

$$TY_{AERO} = QS \cdot C \cdot (MO + MAL \cdot \widehat{ALP} + (MAD \cdot (\dot{ALPHA} - \widehat{QW}) + MQ \cdot \widehat{QO}) \cdot C/(2\ VT) + MU \cdot UP$$
$$+ MDE \cdot \widehat{ELE} + MTR + MSP \cdot \widehat{SPO} + MGE\ KGE + MDS \cdot \widehat{STA} + MB) \cdot KMB$$

2. Evaluate FX1, FZ1 and TY1. Input these values into Equations A-1, A-3 and A-5 respectively.

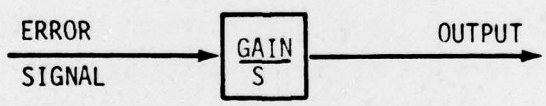
3. Input the relevant trim flight information along with the aerodynamic derivatives.

4. Simplify the equations and solve for XO, ZO and MO.

APPENDIX B
DETERMINATION OF MODEL UNKNOWNS USING INTEGRAL CONTROLLERS AND STEADY STATE

A successful computer simulation of any system requires proper initialization of the computer model. This initialization process is often hampered because there are unknown terms which must be evaluated and input into the model. Determination of these values, however, is not a simple procedure because of the coupling between terms in the model. This appendix describes a modeling procedure which uses pseudo-integral controllers* in conjunction with the EASY STEADY STATE command to determine appropriate values for the model unknowns.

This modeling procedure is based upon the integral controller depicted in the following block diagram.



To implement an integral controller to evaluate any unknown value in the model requires equating the unknown with the controller output, selecting an appropriate error signal and specifying a gain of appropriate magnitude to assure control accuracy. The criteria for determining the error signal and the gain are presented in the following paragraphs.

The integral controller block diagram can also be written in equation form as:

$$\text{OUTPUT} = \int (\text{GAIN*ERROR SIGNAL})dt \qquad \text{B-1}$$

differentiating Equation B-1 gives the rate of change of the output,

---

*The pseudo-integral controller is not a system controller, it is only a mathamatical tool to initialize the aerodynamic model.

$$OUTPUT = GAIN*ERROR\ SIGNAL \qquad B\text{-}2$$

When the STEADY STATE command is executed, the rate of change of the controller output is driven to "zero". Inorder for Equation B-2 to approach zero, the error signal must go to zero. Thus to satisfy Equation B-2 the first criterion for selecting an appropriate error signal requires that the error signal go to zero whenever the model is at the specified initial condition.

The second criterion for selecting an appropriate error signal is the error signal must be coupled to the model unknown (i.e. a variation in the model unknown produces a change in the error signal).

To illustrate these two criteria, let's consider the problem of finding an engine thrust which will maintain the u body axis velocity at 150 ft/sec. An appropriate error signal would be (150 -u). This error signal satisfies both criteria because it goes to zero when u=150 ft/sec, (i.e. the specified initial condition) and the u velocity is coupled to the engine thrust.

The criterion for selecting the gain is based on the STEADY STATE command's definition of a zero rate. EASY defines a steady state to exist whenever the magnitude of the rate is less than .0001. Substituting this criteria into Equation B-2 gives the following inequality:

$$-.0001<(GAIN*ERROR\ SIGNAL)< +.0001 \qquad B\text{-}3$$

Equation B-3 is used for selecting the controller gain. It may not be evident but the magnitude of the gain determines the accuracy of the steady state results.

As an example to illustrate using Equation B-3 for determining an appropriate gain, let's assume that the elevator deflection is unknown. The aircraft pitch is known and must be equal to .1 deg for proper initialization of the

computer model. The error signal for this example is (.1-PITCH). As a first guess let GAIN=.001. Substituting these values into Equation B-3 gives the pitch range which will satisfy the EASY steady state criterion. This range is derived in the following sequence of equations.

$$-.0001<.001*(.1-\text{PITCH})<+.0001$$

$$-.1<(.1-\text{PITCH})<+.1$$

$$-.2<-\text{PITCH}<0.0$$

$$0.0<\text{PITCH}<+.2$$

Thus when GAIN=.001, the STEADY STATE command will control the accuracy of the aircraft pitch to .1±.1 degree. If an gain of 1000 was used the accuracy of the pitch would be specified to .1±.000001. Thus the results from this analysis indicates that the gain should be set to 1000 inorder to assure that the pitch remains at .1 degree while the integral controller sets the elevator deflection.

To summarize the general model implementation procedure for creating a pseudo-interal controller the user must:

- Determine an error signal which goes to zero at the initial condition operating point. Also the variable used in the error signal must be coupled to the model unknown.
- Determine an appropriate gain which assures the accuracy of the results.
- Equate the output of the controller to the model unknown.

In general, multiple integral controllers will be specified, with one controller for each unknown. Once the controllers are implemented, the STEADY STATE command is requested to set the controller outputs to the unknown values in the model. Three examples of the pseudo-integral controller method are given in Sections 9.2.1, 10.2.1 and 11.2.1.

REFERENCES

1. Wahi, M. K., Duleba, G. S., and Perkins, P. R., EASY-ACLS Dynamic Analysis: User's Manual, AFFDL-TR-79-3106, September 1979 (In Press).

2. Aerodynamics Technical Memo 222, "Test Data - Low Speed Wind Tunnel Tests of Jindivik MK 3A Carrying MK 7 Pods", T. H. Trimble and M. K. Politz, Dept. of Supply, Australian Defense Scientific Service Aeronautical Research Laboratories, no date.

3. An Appendix - "Aerodynamic Derivatives and Autopilot/Servo Transfer Functions for MK 203A Short Span Jindivik fitted with MK 4/8 Wing Tip fuel Pods," - Government Aircraft Factories, no date.

4. AFFDL-TR-74-64, Laboratory Tests of Air Cushion Recovery System for the Jindivik Aircraft, Vaughn, J.C.; Steiger, J. T.; Pool, D. J.; and Campbell, S.; April 1974.

5. AFFDL-TR-77-21, Integration of Air Cushion Landing System Technology into the Jindivik RPV, Lloyd, A. J. P.; McAvoy, J. J.; and Rajpaul, V. K.; March 1978.

6. AFIT/GGC/EE/78M-5, An Analysis of the Stability of an Aircraft Equipped with an ACRS, A Thesis Report, Stafford, Max.; March 1978.

7. A Digital Computer Flight Simulation of an Air Cushion Landing System Aircraft, In Publication, Dr. G. Kurylowich, AFFDL.

8. DATCOM Section 4.7, "Ground Effects at Angle of Attack", Revised February 1972.

9. Aircraft Aerodynamics, DeHavilland Aircraft of Canada; Downsview Ontario AEROC 52G2, April 13, 1974.

10. Bell Aerospace Company 1/4 Scale Drawing General Arrangement ACLS/CC-115, Drawing Number 80070 7396-099001.

11. AFFDL-TR-72-4 Part I, Air Cushion Landing System CC-115 Aircraft, A. V. Coles, May 1972.

12. Bell Aerospace Company 1/4 Scale Drawing Hole Installation #2 Flight Trunk ACLS/CC-115, Drawing Number 80070 7396-185074.

13. Air Cushion Landing Systems First Conference December 1972, Compliled by Hrishikesh Saha, Copyright 1973 by University of Tennessee Spaa Institute.

14. T. D. Earl, "Elastically Retracting ACLS Trunks", Article prepared for Canadian Aeronautics and Space Journal, October 1974.

15. ITM/ACLS CC-115/5P/622, United Aircraft of Canada Limited, "ASP-10 Demonstrated Performance".

☆U.S.Government Printing Office: 1980 — 657-084/416